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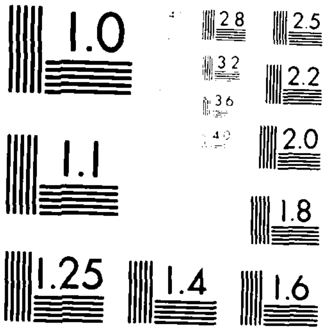

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AD-A152 021

ANALYSIS OF ORBIT TRANSFER VEHICLES

FOR GPS BLOCK 3 SATELLITES

THESIS

David P. Boyarski
Captain, USAF

Stephen P. Mahoney
Captain, USAF

AFIT/GSO/OS/84D-2

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The overall objective of this research was to determine the feasibility and the cost optimum system for using electric OTVs to move Block [illegible] GPS satellites from LEO to a 10,900 km orbit.

For the EOTV, the propulsion systems considered were present and 1990's technology ion engines using mercury, xenon or argon as propellant. There were two power sources examined, a 100 kW nuclear reactor and solar arrays. A system cost model which combined payload, power source, and OTV, and [illegible] launch parameters with algorithms characterizing the electric thrusters was used. The goal was to find the least costly system which had a trip time equal to or less than 90 days. These systems were then compared with the chemical, [illegible], and IUS in terms of total deployment costs for 28 GPS satellites launched at a rate of four per year for seven years.

The studies found that a reusable EOTV with [illegible] mercury ion engines powered by gallium arsenide concentrator arrays could perform the mission for [illegible] of the cost of the cheapest chemical system. The nuclear powered OTV, while less costly than the chemical systems, was not as competitive as the solar EOTV. The weight of the nuclear reactor and its heat radiators required the use of 37 engines resulting in higher costs for the system.

AFIT/GSO/OS/84D-2

ANALYSIS OF ORBIT TRANSFER VEHICLES FOR GPS BLOCK 3 SATELLITES

THESIS

Presented to the Faculty of the School of Engineering
of the Air Force Institute of Technology
Air University
In Partial Fulfillment of the
Requirements for the Degree of
Master of Science in Space Operations

David P. Boyarski
Captain, USAF

Stephen P. Mahoney
Captain, USAF

December 1984

Approved for public release; distribution unlimited

Preface

The intent of this study was to take a look at using electric Orbit Transfer Vehicles for deploying GPS satellites from a low earth orbit to their destination orbit, and to compare them with chemical OTVs for the same mission. It was felt that the use of electric systems would produce tremendous cost savings over the chemical systems presently used.

Since there would be two individuals working on this thesis, a natural division point was the type of power source used by the electric system. The two considered here were nuclear and solar. The initial background work was performed together, the power analyses performed seperately, and the final cost analysis, sensitivity, and conclusion were a joint effort.

The cost figures used in this paper are based upon estimations made by experts in their respective areas. Though as accurate as possible, it must be understood that as time unfolds these numbers may change.

We gratefully acknowledge the assistance and advice of several individuals without whose help this thesis would not have been possible. They include our thesis committee, LTC (Dr.) Mark Mekaru and Mr. Dave Massie, and two experts in the fields of space nuclear reactors and solar power, Mr. Dave Buden and Dr. Pat Rahilly. Finally, we both wish to thank our wives, Jane and Deanne, for their tremendous patience and understanding during this thesis effort.

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ABSTRACT

The overall objective of this research was to determine the feasibility and the cost optimum system for using electric OTVs to move Block 3 GPS satellites from LEO to a 10,900 nm orbit.

For the EOTV, the propulsion systems considered were present and 1990's technology ion engines using mercury, xenon or argon for a propellant. There were two power sources evaluated, a 100 KW nuclear reactor and solar arrays. A systems cost model which combines payload, power source, trajectory, and earth-to-LEO launch parameters with algorithms characterizing the electric propulsion system was used. The goal was to find the least costly systems which had a triptime equal to or less than 90 days. These systems were than compared with the PAM D-II, CENTAUR-G, and IUS in terms of total deployment costs for 28 GPS satellites launched at a rate of four per year for seven years.

The studies found that a reusable EOTV with 12 mercury ion engines powered by galium arsenide concentrator arrays could perform the mission for 42% of the cost of the cheapest chemical system. The nuclear powered EOTV, while less costly than the chemical systems, was not as competitive as the solar EOTV. The weight of the nuclear reactor and its heat radiators required the use of 37 engines resulting in higher costs for the system.

ANALYSIS OF ORBIT TRANSFER VEHICLES

FOR GPS BLOCK 3 SATELLITES

CHAPTER I. BACKGROUND

INTRODUCTION

No longer are large booster rockets the only means to launch a satellite into outer space. The shuttle provides routine access to low earth orbits and offers a variety of options for transferring satellites to higher orbits. To date orbit transfer vehicles (OTV's) have been limited to single use chemical bipropellants and solid rocket motors (SRM's). However, a simple, reuseable, orbit transfer vehicle could tremendously reduce launch costs (38, 40, 43, 69).

Within the last year, electric OTV's (EOTV's) have begun to receive greater attention for orbit transfer missions (12). They possess several desirable features which make them viable options to the present systems. EOTV's have low fuel consumption, high specific impulse, and low acceleration (32, 38, 43, 76). The potential cost savings obtained from reusable electric OTV's has yet to be fully assessed. However, it is felt that cost savings will be realized from the low fuel consumption of the electric OTV's (14, 28, 40, 48).

Most studies concerning electric OTV's consider only

heavy payloads being placed in geosynchronous orbit (GEO) (30, 40, 48, 51, 69). Few studies found by the authors consider transporting light payloads to GEO, and none considered transferring a light payload to a mid-orbit.

Captain Sponable at Space Division requested an analysis of OTV's for Block 3 GPS satellite deployment. He wanted to determine the applicability of electric OTV's for GPS deployment, including the feasibility, costs involved, and possible scenarios.

PROBLEM STATEMENT

The overall objective of this research is to determine the feasibility and the cost optimum system for using electric OTVs to move Block 3 GPS satellites from Low Earth Orbit (LEO) to a 10,900 nautical mile orbit and to compare it with chemical OTVs. Specific subobjectives are:

1. Analyze power sources
   a. Solar array degradation caused by Van Allen Belt radiation
   b. Solar array size requirements
   c. Problems associated with using nuclear power plants as power sources
   d. Feasibility of these power sources, especially in terms of total vehicle weight
2. Determine the best electric propulsion system in order to meet system demands
   a. Calculate transfer times for the various systems using formula or Alfano-Weisel curves
   b. Insure that the system will meet user's constraints and will be feasible in the 1990 time frame

3. Conduct a cost analysis

   a. Assign costs to vehicle, launch, maintenance, and ground tracking support

   b. Compare overall costs for deployment of 28 satellites (4 replacements per year for 7 years)

<u>METHODOLOGY</u>

Two separate but similar methodologies were used for this research problem. The problems and constraints inherent in a solar powered electric orbital transfer vehicle (SOTV) differ greatly from those of a nuclear powered OTV (NOTV). The general methodology is discussed below, with specific approaches for nuclear and solar designs presented in future chapters.

First, a data base will be compiled from the literature search. Specifically, the values for specific impulse, engine weight, satellite weight and other relevant parameters will be compiled. Also, data concerning the solar array degradation will be compiled and confirmed with solar array experts. Similar parameters and operating characteristics will be found for a 100KW nuclear reactor.

While a detailed design of the electric propulsion system will not be undertaken, the design will consider feasibility, shuttle adaptability and useage, payload protection, and power source. The power source will be the main subsystem analyzed. A nuclear reactor power source will greatly affect the system design because of its large mass

and radioactivity. For the solar powered system, the size of the power source will be of principle concern.

Using the above data, the different systems will be evaluated using a limited cost analysis. Costs will be established for important system components and support measures and will be applied to each system model. Based on these costs, the optimal system will be chosen.

This thesis will be a two man effort. Individual efforts will be undertaken in the evaluation of EOTV's based on their power source. Captain Mahoney will examine a design using a 100KW nuclear generator as the power source and Captain Boyarski will evaluate the use of solar arrays. The initial data gathering and the subsequent comparison of EOTV's to other transfer vehicles will be undertaken jointly.

CHAPTER II. LITERATURE REVIEW

SCOPE

Before undertaking any evaluation of EOTVs, there are some background questions that must be answered:

1) What studies, if any, have been made concerning the use of EOTVs and what were their findings?

2) What electric rocket engines are either operational now or forecast to be by the 1990s (the planned deployment time of the Block 3 satellites) ?

3) What options are available for use as a power supply and what problems, if any, are associated with each option ?

The purpose of this literature review is to determine if there is sufficient reason to commit the time and resources to an evaluation of EOTVs for the GPS Block 3 satellite deployment.

I. What studies, if any, have been made concerning the use of EOTVs and what were their findings?

There are many studies that address the use of EOTVs (18,26,27,28,29,32,40,43,48,52,59,68,69,70,74,84). Most address the transfer of spacecraft from LEO to GEO (Geosynchronous Orbit - 19,300 nautical miles) and they all indicate that reusable EOTVs are not only feasible but are extremely cost effective, especially for larger payloads. All reports point out that the electric rocket engines on

EOTVs have a much higher specific impulse than chemical rockets and would require less propellant to perform the same mission. This smaller fuel requirement, besides being a benefit in itself, reduces the payload requirement of the earth-to-orbit (ETO) launch vehicle.

Individually, some of the reports contained other relevant findings. Mr. D. G. Fearn (28) found that for payloads of 1000 kilograms (2200 pounds) or less, it is most economical to use dedicated EOTVs while for larger payloads, a reusable EOTV would be best. In a later report (27), Mr. Fearn estimates that 25% of the cost of a satellite system would be attributed to transportation. Reusable EOTVs would reduce the number of needed launches by over 50% and thus greatly reduce overall transportation costs.

Captain Lee Maddox, in his thesis (52), found that in terms of dollars per kilogram payload delivered, EOTVs had the lowest Life Cycle Costs. The reusable bipropellant system, although reusable, will be a very costly system because of the weight of the fuel required. An extra shuttle flight will be necessary to refuel the system each time it is used. Captain Maddox also presented a methodology to determine the number of EOTVs necessary to maintain a given deployment schedule and to evaluate the cost of such a system.

Captain David Perkins (59) compared four different propulsion concepts: solid rockets, cryogenic systems, solar rockets, and electric propulsion engines. He found that for

a total initial weight (payload plus propulsion unit) of 85,000 pounds, the electric propulsion engines could deliver 65,000 pounds to GEO. The other systems could deliver 5,000 pounds, 23,000 pounds, and 45,000 pounds, respectively.

Similarly, Mr. J. Rehder (69,79) performed a preliminary cost analysis on OTVs and fleet sizes and found that EOTVs were less costly than chemical systems.

Captain Jess Sponable, in his thesis (74), found that in order to optimize the Space Transportation System, it was necessary to deploy a space station in low earth orbit (LEO) and develop a reusable orbit transfer vehicle (OTV). In subsequent conversations with Captain Sponable, he indicated that although a space station will not be deployed for a few more years, a reusable OTV by itself should still be able to reduce overall costs for satellite deployment. This would result from savings in the earth to LEO portion of the deployment and from the savings in purchasing fewer upper stages.

Boeing Aerospace Company conducted a study to identify those areas in the field of electric propulsion technology where advances in the state-of-the-art are required to allow developement of propulsion systems for the future (26). In this study, they presented a system cost model in which certain parameters representing the payload, the power source, the trajectory, and the earth-to-low-orbit launch system were combined with algorithms characterizing the electric propulsion system. The model produced a set of

costs for each of the missions it considered. While the algorithms used include important factors such as earth shadowing and solar array degradation, the report did not explain how the values were determined. Conversations with one of the authors revealed that most values used were only the best guess of those performing the study. Therefore, while the system cost model is good, there is room for improvement in some areas of parameter value determination.

One major finding of this study was that for mercury ion thrusters, a specific impulse of 3000 seconds was optimum for most missions. This was also the major finding of a similar study performed by Regetz and Terwilliger (68).

Mr. R. M. Jones compared only the performance of present day electric propulsion systems for orbit transfer and found that mercury and xenon ion thrusters were best for high values of specific impulse (40). He states that besides thruster efficiency, a low specific mass power supply is the most important factor in electric propulsion.

While it may appear as though EOTVs are without drawbacks, this is not the case. All of the reports agree that the major drawback of EOTVs is that they are slow. Because of the low thrust levels inherent in electric rocket engines, the transfer time from LEO to GEO and back can take from 100 to over 350 days. The long transfer time means that these systems are not suitable for priority cargo nor are they suitable for manned vehicles. In addition, the long transfer time means that passage through the Van Allen belts

will be slow, and the vehicle will be subject to impact by high energy particles for several days. This will cause degradation of solar panels ,if used, and will require that the satellite have extra shielding and thus weigh more (27,28,32,48).

Despite the problems with EOTVs, their reusability and their capability to move large payloads make them economically attractive (28). Although none of the studies specifically address the orbit transfer for the GPS Block 3 satellite, the methodologies used in the studies, in particular those in References 1, 42, and 68, provide a means of analyzing the mission in question. None of the studies specifically said nor implied that EOTVs could not be used economically for satellite transfers from LEO to intermediate height orbits. This area is still open for further investigation.

II. What electric rocket engines are either operational now or forecast to be by the 1990s ?

There are many reports that evaluate the different types and variations of electric rocket engines. However, many of the engines evaluated are only in the conceptual or laboratory stage and are many years away from being operational. Therefore, the search for data was limited to those engine types that have already been tested and used, or are forecast to be operational by the 1990s. The only engines initially found to be in this category were the

Electron Bombardment Ion Thruster (more commonly referred to as the ion engine) and the Arcjet. Basic explanations of the principles of operations of these and other electric engines are available in most texts on rocket propulsion. References 38 and 52 are particularly good and very understandable.

The only electric engine that has been tested in space is the ion engine. Two reports on the results of SERT II (Space Electric Rocket Test II) provide extensive data on the performance of ion engines aboard the spacecraft (45,47). The SERT II spacecraft was launched in 1970 on a one year mission as a test bed for ion engines. It continued to operate until May 1981 when the engines finally ran out of fuel. It successfully demonstrated 300 restarts of the ion engine, and one engine operated for nearly 10,000 hours (14 months) in space (47). The spacecraft also demonstrated the capability to throttle the engine - that is, to operate it at various power settings.

Captain Maddox (52) performed an extensive evaluation on 1990 technology variations of the 30 cm Kaufman thruster (an ion engine) and found that the Ring-Cusp 3-Grid 30 cm configuration using xenon propellant was the best option for this type of engine. It was the engine that weighed the least for a given thrust. Data on this engine is available in his thesis.

Information on the performance capabilities of present generation ion thrusters can be found in many sources (2,9,10,18,36,40,42,53,62,64,67). Most of these sources also

contain projected performance levels for the 1990s.

During briefings on EP systems which the authors attended at the NASA-Lewis Research Center, information on the status of many electric thrusters was presented. From these briefings and subsequent conversations with the briefers, it was decided that while much work is being done on arcjets and resistojets, these technologies will not be developed enough for use in the mission being examined during the 1990's (66).

III. What options are available for use as a power supply and what problems, if any, are associated with each option ?

Because electric rocket engine performance capability is limited mainly by available power, the most frequently mentioned problem with EOTVs is that of finding an adequate power supply. Possible power supplies are solar arrays and nuclear generators.

In the area of solar arrays, the most serious problem mentioned in the reports was that of solar array degradation due to Van Allen belt radiation. Three of the reports estimated that there would be 40-50% degradation of the solar arrays (27,32,48); however, they did not explain how these estimates were derived. Only one report (27) addressed the issue of solar array protection options to decrease the degradation and only one option did not substantially add to the weight of the solar arrays. This option, which involves a thermal annealing process, is still being tested (21).

This problem of solar array degradation caused some engineers (27,69) to question the reusability of the EOTVs. Furthermore, the cost of an EOTV is dominated by the cost of the power generation system; therefore, to remain economical, a higher degree of reusability of the power generation system is required (69).

The second problem with solar arrays is that of size and weight. In their paper on "Future Military Space Power Systems and Technology," Mr. Barthelemy and Mr. Massie forecast substantial improvements in both areas by 1990. Technologies being examined by the Air Force Advanced Light Weight Solar Array Blanket program are predicted to produce arrays with a weight decrease of 4 to 1, and a size decrease of 2 to 1, while increasing hardness to radiation (6).

Two sources (1,57) present what can be expected in the next generation of solar arrays. Not only do they contain information on specific mass and specific power, they also include information on the structures needed to support the arrays, their weight, packaging, and deployment methods. The arrays discussed are derivatives of the array deployed on the initial flight of the shuttle Discovery.

As far as using nuclear generators as a power supply is concerned, the first thing most people ask is not is it technically feasible but is it safe and legal. This question is addressed in several sources (20,33,34,39,77,82) and the consensus is that it is safe and legal.

The general design and workings of nuclear reactors are

covered in books by Loftness (49) and by Dietrich (23). The components that make up the reactor as well as the actual power conversion methods and units are discussed.

Information specifically on space based nuclear generator systems is contained in several articles (2,15,16,42,54,56). These mainly address the SP-100 reactor program, its goals, performance capabilities, and present status. The goal of the program is to design a nuclear reactor to provide 100 kilowatts (Kw) of power. Some of these articles (15,16,42,56) also contain methodologies for analyzing the use of nuclear generators as a power source for EOTVs. One common finding from the studies is that nuclear generators are best suited for use with heavy payloads or for long duration missions, especially those that travel away from the sun. In order to compensate for the high specific mass of the reactor, the mission must take advantage of the nuclear reactor's strong points - its capability to provide continuous power for a period of several years.

CONCLUSION

As a result of the literature review, it was found that no studies had been conducted on using EOTVs for the transfer of satellites from LEO to intermediate height orbits. Methodologies to perform such a study do exist for both solar arrays and nuclear generators as power sources. Further investigation into determining proper values for solar array

degradation due to Van Allen belt radiation is needed. Likewise, the effect of the earth's shadow on solar powered EOTVs needs to be defined.

The only electric propulsion technology that appears to be available for use in the timeframe being considered is the ion thruster. Possible propellants for this thruster are mercury, xenon and argon. Sufficient information on present and expected performance levels for these engines exists.

Since there are no studies that say using EOTVs for this mission is not feasible and since the data and methodologies to perform the study are available, it is felt that further investigation is warranted.

CHAPTER III. NUCLEAR POWER ANALYSIS

INTRODUCTION

Nuclear reactors represent a source of great power both on earth and for space applications. The consideration of using a nuclear power source (NPS) for powering an OTV stems from its ability to produce a large amount of power, its small size, and its relatively safe useage (14, 15, 16, 20, 33, 34, 39, 54, 82). The methodology for NPS is almost identical to that used for solar power. The methodology will be presented by first describing the research problem in terms of using a nuclear power source. Next, the specific assumptions made for this analysis will be presented. This will be followed by describing the scope of the research. Then the method and equations used to perform the analysis will be considered, resulting in the final Basic program which was used. These results will then be examined, the optimum choice for an OTV selected, and a limited cost analysis made using these figures.

PROBLEM STATEMENT

The specific objective of this portion of the research is to find, using the number of engines as the independent variable, the minimum cost system for operating an EOTV using a nuclear reactor power plant as the source of energy. Then, provided that this system meets all of the feasibility constraints imposed upon it, such as orbit transfer time, shuttle adaptability, and cost, it will be the system used in

the cost analysis. If the minimum cost system does not meet these feasibility constraints, then the number of engines will be increased or decreased, trading cost for another parameter, until the system meets the user's needs.

ASSUMPTIONS

Several assumptions were made in performing this analysis. In particular, it was assumed that the SP-100 program reactor (100 KW, pictured below) will be developed on schedule and therefore will be available by the 1990's.

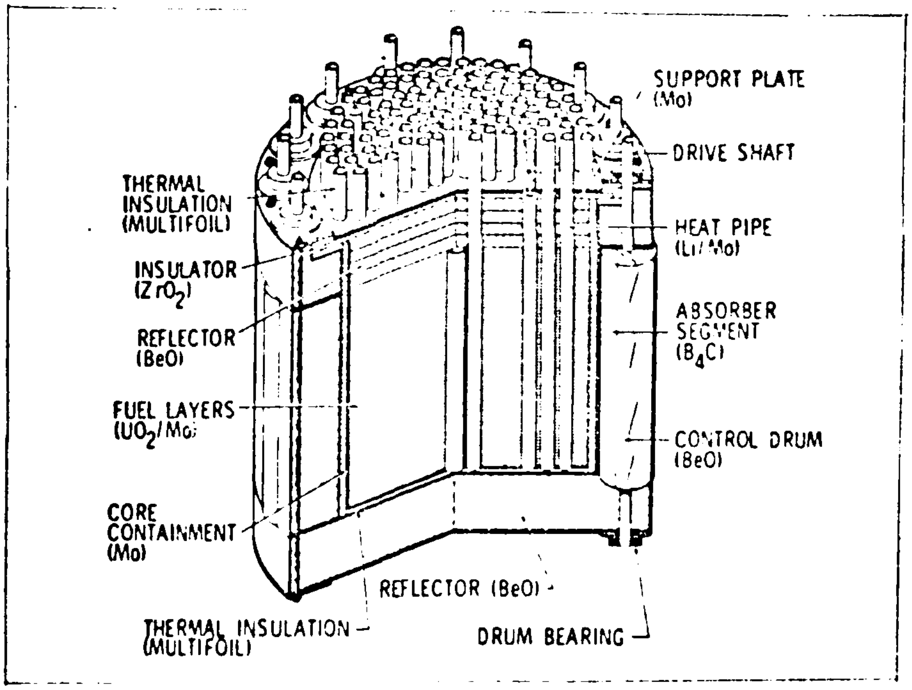


Figure 3-1. 100 Kw Space Nuclear Reactor (56)

Along with this technology, it was also assumed that the electric engine technology will be as predicted for that time frame. The continuing requirement for GPS satellites or an equivalent system was assumed, and it appears reasonable

considering the increase in civilian requests for GPS use. No spare engines will be carried with the electric systems, and it is assumed that any trip time penalty resulting from a partial failure of the system is acceptable. One final assumption is that the shuttle will be available for launching these electric OTV's, and considering the estimated launch rate of four per year (75), this also appears to be well founded.

SCOPE

In determining which systems to consider for this analysis, the problem had to be scoped to a reasonable level. Background information showed that for the electric systems, only the arcjet and ion type engines were feasible alternatives (2, 12, 66). Further discussion at NASA-Lewis

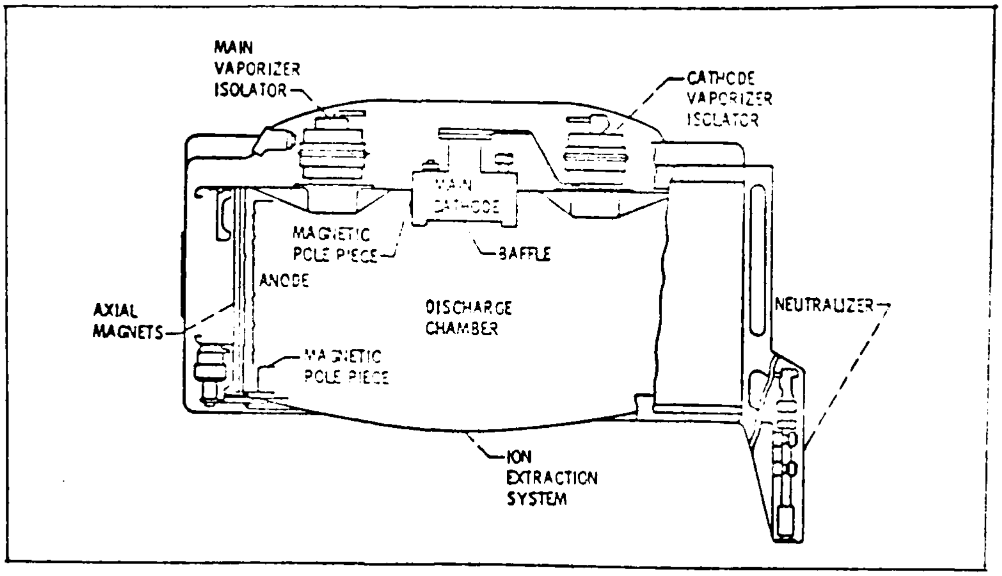


Figure 3-2. Ion Engine Schematic (52)

revealed that of these, only the ion engine is being seriously considered for large scale future use. Therefore, this is the only electric engine considered in this analysis (17, 66). In addition, hybrid chemical and electrical systems were not considered.

Analyzing only one satellite mission, that of GPS, allowed for a constant payload weight (1500 Kg) and orbit change (LEO to one-half GEO, 55 degree inclination). The only chemical OTV's considered for comparison were the PAM-D II, Centaur-G, and IUS. The SP-100 reactor was assumed to have a maximum operating capability of 100 kilowatts because of the time frame in which it is used. It will be scalable to much higher energy levels in later years. The reactor and OTV are designed so that the radiation levels emitted by the reactor are acceptable for this mission. This left only the Van Allen belt and solar radiation to be considered. The cost analysis was limited to the three areas of hardware, launch, and operations.

METHOD AND EQUATIONS

In designing a methodology to use for the nuclear powered system, the sizing procedure shown below was used. Most of the parameters were constants, but a few, notably trip time and total system mass, were not. Some of the parameters used in this model, such as occultation, were not applicable to the nuclear powered system, and were therefore set equal to zero.

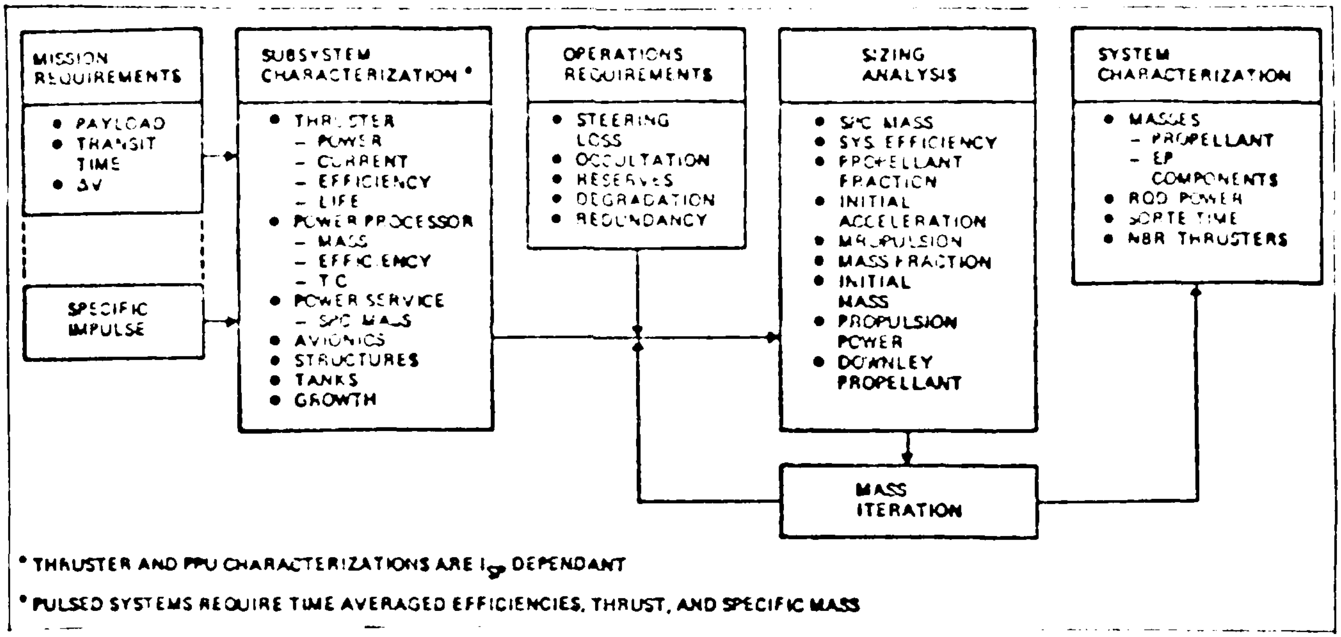


Figure 3-3. OTV Sizing Procedure (2)

In developing a set of equations for this project, the easiest method appeared to be to divide the OTV structure and costs into simple, separate areas. This was later found to be the technique used by most other researchers as well (2, 15, 42). The OTV was divided into three main areas listed below.

PROPULSION SYSTEM	POWER SYSTEM	OTHER STRUCTURES
Engines	Reactor	Boom
Power processors	Radiator & tubes	Van Allen Belt protection
Fuel and tanks	Thermoelectric devices	Satellite-Shuttle adapter
Associated structural hardware	Shielding	
	Pumps, working fluid	

In most cases it was possible to derive a single number relating to cost, mass, etc., which accurately represented each OTV area. A simple design configuration is shown below with a full discussion of the system design included in

Appendix B.

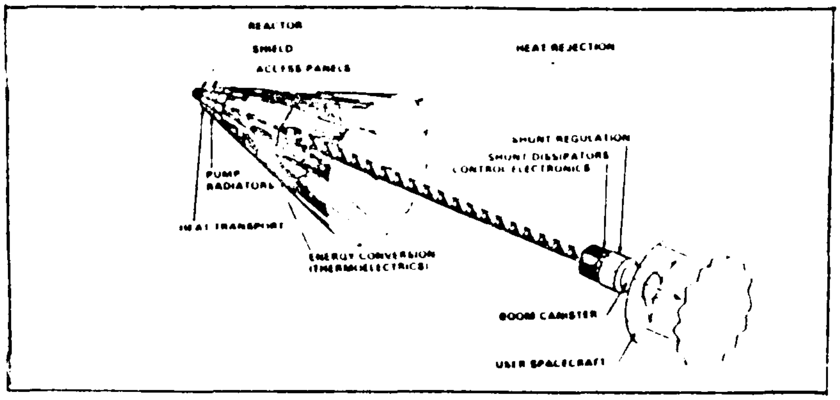


Figure 3-4. A Simple Nuclear OTV Design (15)

The costs were separated into the 3 areas shown here.

LAUNCH COST	OPERATIONS COST	HARDWARE COST
Launch cost for shuttle (based on either mass or size)	Cost of tracking and control operations	Structure cost Power system cost Engine system cost

Most of the actual OTV equations, other than the cost equations, were taken from several sources and confirmed by checking them against one another. They were also compared with the equations used in the solar analysis and found to be equivalent. A full discussion of these basic rocket equations will not be performed here, but may be found in any good propulsion book. An entire listing of the equations used is included in Appendix A, and a discussion of the relevant equations can be found in Appendix C. Several equations do require attention and explanation here as they drive the entire process. The first is the total cost equation.

Total cost = Launch cost + Hardware cost

+ Operations cost (3-1)

The individual costs in the above equation include all of the costs for the acquisition and operation of the OTV system. Further defining these separate costs:

Launch cost = ((OTV Length (ft)/60)/.75) * 65,000,000 (3-2)

Hardware cost = sum of (propulsion system, power system, other structures) (3-3)

Operations cost = trip time * ops cost per unit time (3-4)

Because everything in this analysis has been based in terms of cost, the above equations nicely divide the entire system into manageable parts for analysis. The derivation of the launch cost equation is shown in Appendix C, per NASA pricing regulations.

One equation which deserves special attention is the trip time equation. Due to the equations used, the only variables which cannot be directly determined are the trip time, total fuel, and total mass. These must be arrived at by using an iterative process. A simple schematic is shown below.

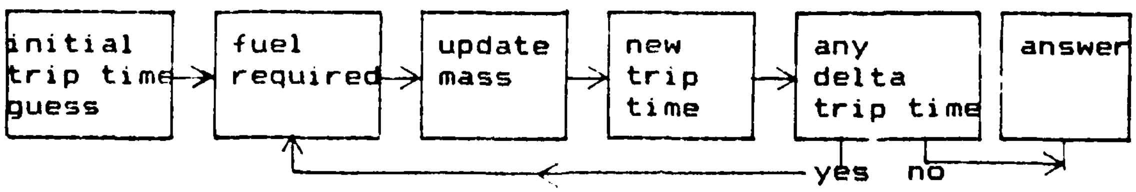


Figure 3-5. Iterative Process to Determine Trip Time

The trip time is determined by the equation (42)

$$TT = \frac{(MoVo)}{(NT)} \left(1 - 2 \frac{Vf}{Vo} \cos(1.414\ i) + \left| \frac{Vf}{Vo} \right|^2 \right)^{(1/2)} \qquad (3\text{-}5)$$

where

TT=trip time (one way) (sec)
Vo=initial orbit velocity (m/s)
Vf=final orbit velocity (m/s)
i= inclination change (rad)
T=thrust (N)
N=# of engines
Mo=launch mass = f(fuel mass) = f(tt) = f(launch mass) (Kg)

As illustrated by the schematic, a transcendent relationship exists. The fuel required must be determined. But this is dependent upon the trip time, which is again dependent upon the launch mass, and therefore the fuel required. Once the fuel required is fully determined, the rest of the calculations can be performed and the costs calculated.

BASIC PROGRAM

Before deciding to use a simple Basic program run on an IBM home computer, several other methods had been tried. The first attempt was the use of a Fortran program called Process, a Multi-Objective Non-Linear Optimization routine. After several weeks of attempted use, it was decided that this was not the proper program to use for a single objective problem, although in some cases one may be able to convert it for this purpose. A Single Objective Optimization program, SUMT (Sequential Unconstrained Minimization Technique), was sought but found not to exist on the computer. In fact,

Process was simply a modified SUMT program, and this was why so much time had been spent on Process initially. A remodification attempt on Process to once again place SUMT on the computer failed because of a time restriction and computer problems.

Next, an optimization routine called MPOS (Multi-Purpose Optimization System) was used and found acceptable for obtaining test results. However, as the problem expanded, it was discovered that each run would require a new set of equations. This was due to the fact that MPOS is a linear optimization routine only. There is one quadratic routine on MPOS, but no other non-linear routines. This meant that each run required a modified set of equations to remove the non-linear relationships. Instead of iterating, it would be necessary to assume a specific number of engines, remove the nonlinear relationships, and then make one run. For a different number of engines, the process would have to be repeated.

As a result, a Basic program was used which, by doing the process iteratively, allowed the data for each number of engines to be processed. This allows the user to choose a non-optimum cost point at which to operate if, for instance, the trip time at the optimum cost point is unacceptable. It also was excellent in preparing the data bases from which graphs could be made using an available plotting routine.

The program, as indicated above, performs a simple iteration using the number of engines as the iteration

variable. A dummy variable is used to iterate on the trip time and fuel required during each engine iteration. This inner loop was run twenty times for this program and inspection of the runs showed that this loop generally determined the fuel required and trip time to its final value after only 10 iterations. Storing the best costs and writing the results of each engine iteration to a data file gives the minimum cost point in terms of number of engines and creates the graphics data file. Changes are very easy to make, and although this program uses constants which are set within the program, an interactive approach could also be developed with minor modification.

The program is divided into three sections. The first section sets all initial engine parameters. The second section sets the orbit parameters and determines the delta velocity required for orbit transfer. The third section includes the OTV and satellite constants and performs the iterative procedure. A flow chart for the program is depicted in Figure 3-6.

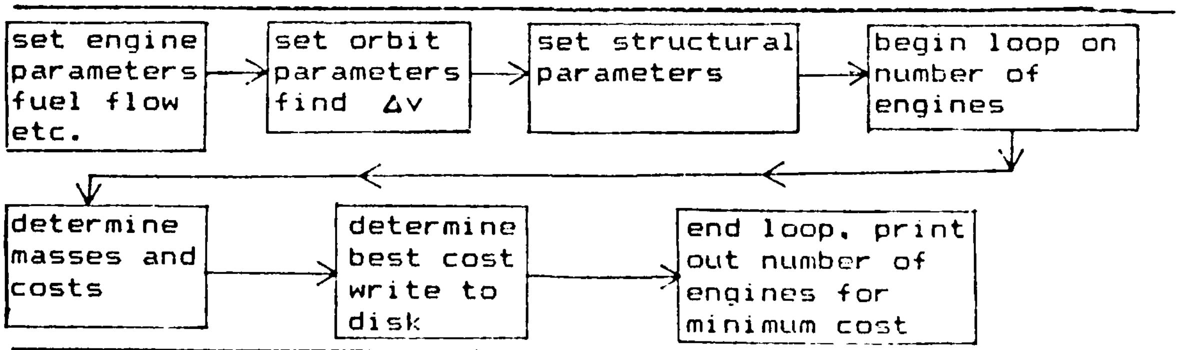


Figure 3-6. Program Flow Chart

The outputs of each iteration which were printed as data files were as follows:

data1 # eng trip time out trip time back total time

data2 # eng launch cost operations cost total cost

data3 total trip time total cost

Finally, the number of engines, total trip time, and total cost for the cost optimum point were printed to the user's screen.

PERFORMANCE ANALYSIS

From an analysis of Hg, Ar, and Xe systems using 1984 capabilities, the best fuel option was determined. The engine and reactor system capabilities were then updated to reflect the standards expected for 1995. Runs were again made to insure that the best fuel option had not changed. From this optimum fuel, 1995 technology system, the minimum cost point was found. The data from this run allowed the user to determine if this minimum point was acceptable. The user could also select a nonoptimum cost point if a different transfer time was preferred due to operational constraints. The user's selection was then used as the data point for computing the resultant costs and trip times for comparison with the solar powered system and the chemical OTV's.

COST ANALYSIS

The cost analysis was used to determine if electric OTV's were competitive with present chemical OTV's. The

costs were limited to the procurement, launching, and use of the EOTV, as previously shown by the cost equations. The analysis considered 28 round trip missions. The cost for this was then compared with the cost for 28 missions using each of the chemical OTV's. It was assumed that there would be no failures during any mission because the reliability data on some of the OTV's is nonexistant, and for the others it is unavailable.

FINAL CONSIDERATIONS

Before we consider the results of this methodology, it is prudent to consider the limitations of this analysis and what the results will indicate. It must be remembered that the systems costs are being determined with figures which are only best estimates and therefore will have some uncertainties in them. It appears that cost figures are closely guarded entities, and accessability to them is hard to obtain. Also, as mentioned previously, some of the decisions concerning these systems have not yet been made, making cost figures hard to determine.

In choosing the cost optimum point for each system, it must be remembered that all this represents is the cost minimum point. It does not indicate that one engine is necessarily better or more reliable than another, nor does it indicate the values of the other system parameters. This leads us to the consideration of what to do if this point is unnacceptable in terms of another parameter. For instance,

if the OTV system mass is beyond shuttle capability, then the cost becomes irrelevant. In this case, an adjustment will have to be made to move away from the cost minimum point in order to make the system feasible, and a tradeoff will occur.

CHAPTER IV. NUCLEAR POWER RESULTS

The nuclear results will be presented in the same manner as the methodology. First, the constant values and engine parameters will be described. The results of the first runs using the 1984 characteristics will then be discussed. Next, 1995 runs will be presented in the same fashion with consideration given to any improvements or options available. Finally, the numbers chosen for comparison with the solar powered system results will be presented.

There were several constants whose values did not change for any of the analysis runs. These are presented below.

Power processor efficiency = .9
Thrust = .129 (N)
Mu earth = 398603.2 (Km^3/sec^2)
Radius earth = 6378.165 Km
Initial orbit radius = 200 Km
Final orbit radius = 20,186.81 Km
Total length of structure = 30 feet
Drag coefficient = .0001
Boom structure mass = 150 Kg
Boom structure cost = 1000 $/Kg
Guidance, navigation and control mass = 50 Kg
Guidance, navigation and control cost = 1,000,000 $
Fuel cost = 15 $/kg (for all fuels)
Ops cost = 10 million $ (varied 5 - 20 for sensitivity)

The thrust was kept at a constant .129 newtons for two reasons. First, it gives a common basis for comparing all the engines. Also, the expected improvements in the 1995 engines are based primarily in terms of thrust. This allows for easy transformation of the future engine parameters. The total length of the structure is determined by the length of the reactor and radiator. Any additional structure will have to fit within this area. The drag coefficient is derived from

the Boeing study (2) and the estimated value is based on the cross sectional area of the NOTV. Without performing a full analysis on structure drag, this has been stated as a reasonable value (17, 66). It has been scaled to the drag coefficient for the solar analysis for the comparison between them.

The structure and guidance masses and costs are estimates but discussion with several experts in the field have shown these to be reasonable values (42, 66). The fuel cost was also kept constant for the three types of fuel considered. This was done for several reasons. First, cost figures for the other fuels could not be readily obtained, and second, the impact of the fuel cost on the total cost is extremely small, less than .1 per cent. The operations cost was set at an estimated 10 million dollars per year to account for numerous increases from the 5 million dollar figure used by some studies (2, 42). It was varied between the values indicated, but only as a sensitivity measure.

The table below represents parameters which were obtained from several sources (2, 15, 40, 43, 47, 66) and used for the 1984 initial runs. These are the same figures as those used in the solar powered analysis.

	Hg	Ar	Xe
Power Processor eff.	.90	.90	.90
Thruster effic.	.67	.755	.80
Isp (sec)	2900	6270	4560
T (Newtons)	.129	.129	.129
Power reqd (Kw)	3.06	5.785	3.98
Engine mass (Kg)	51	55	51
Cost/Engine System ($/Kg)	13,500	13,500	13,500

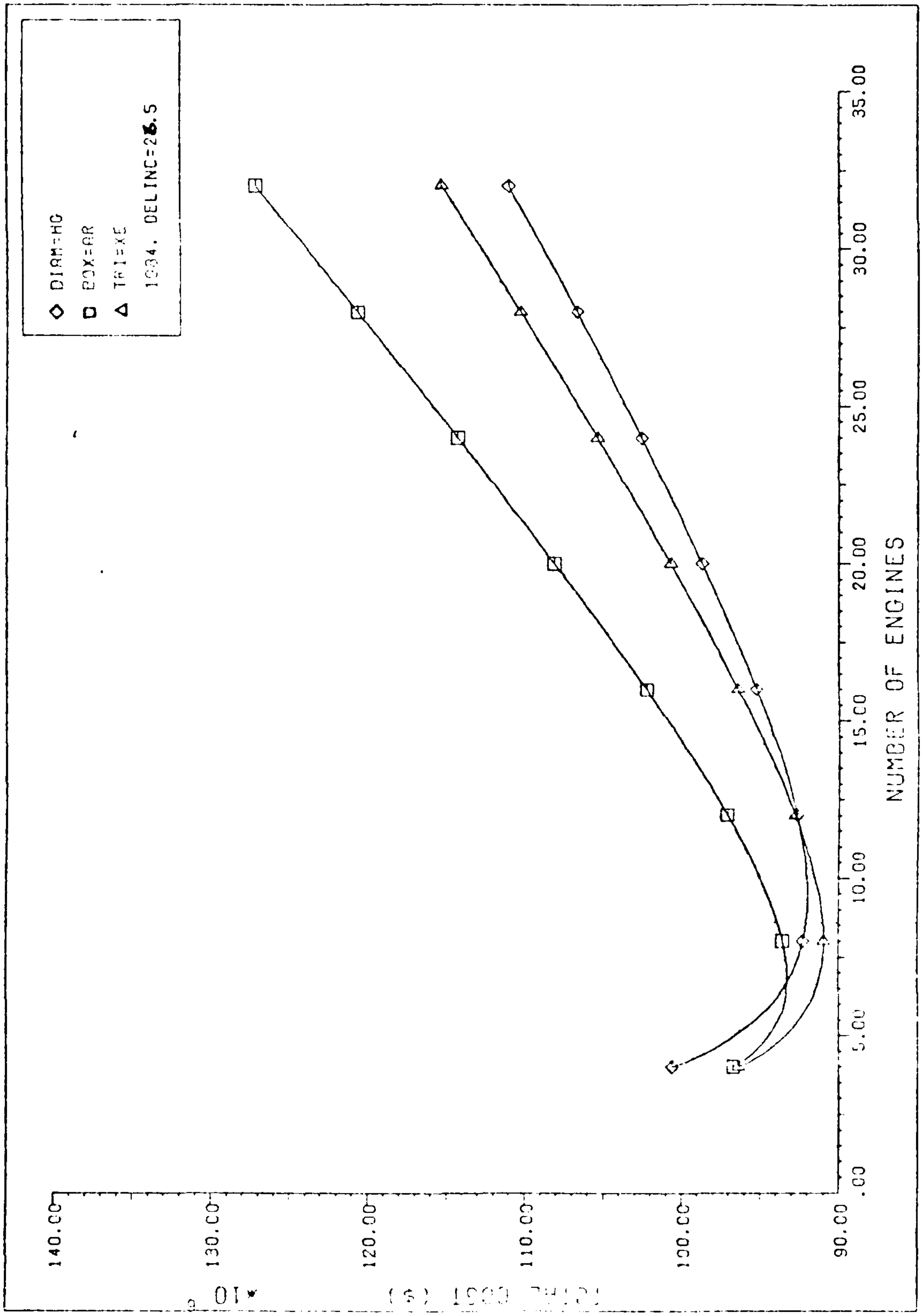
The power required is determined by the equations used in the program and is presented here only for reference.

The results of the initial runs are shown on the graphs in Figures 4-1 thru 4-10. The runs were made using a delta inclination (delinc) of 26.5 degrees. This indicates that the OTV was placed in the standard shuttle inclination of 28.5 degrees and therefore has a 26.5 degree plane change to arrive at the proper 55 degree orbit inclination. A second inclination option will be considered later. A list of the important results is shown below.

1984 DELINC = 26.5

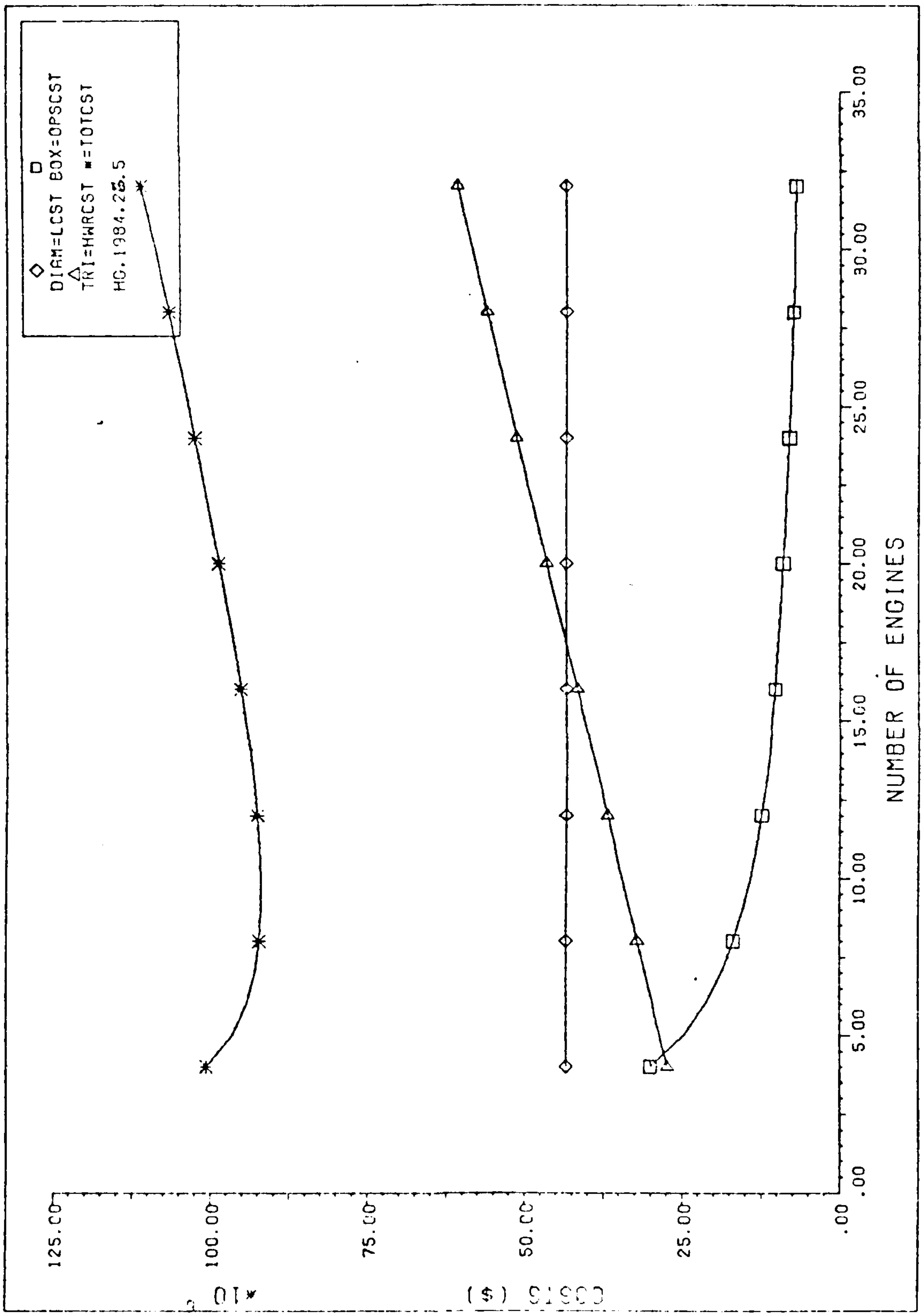
	Hg	Ar	Xe
Total cost (million $)	91.96464	93.25419	90.94012
Total trip time (days)	559.14	561.42	519.17
Number of engines at optimum point	9	7	8

These results show that the Xenon fueled system is the best system in terms of minimum cost. Its trip time is also 40 days less. The user had initially stated that he wished to have an outward trip time of approximately 90 days, with a total trip time of approximately 160 days. It can be seen that the trip times in all cases are unaccptable. Evaluation of the trip time equation and the following discussion describes why these times are so large. The fuel required for each of the different fuel sources varies dramatically. The Mercury, Argon, and Xenon systems require 1973 Kg, 712 Kg, and 1035 Kg of fuel respectively. This, when combined



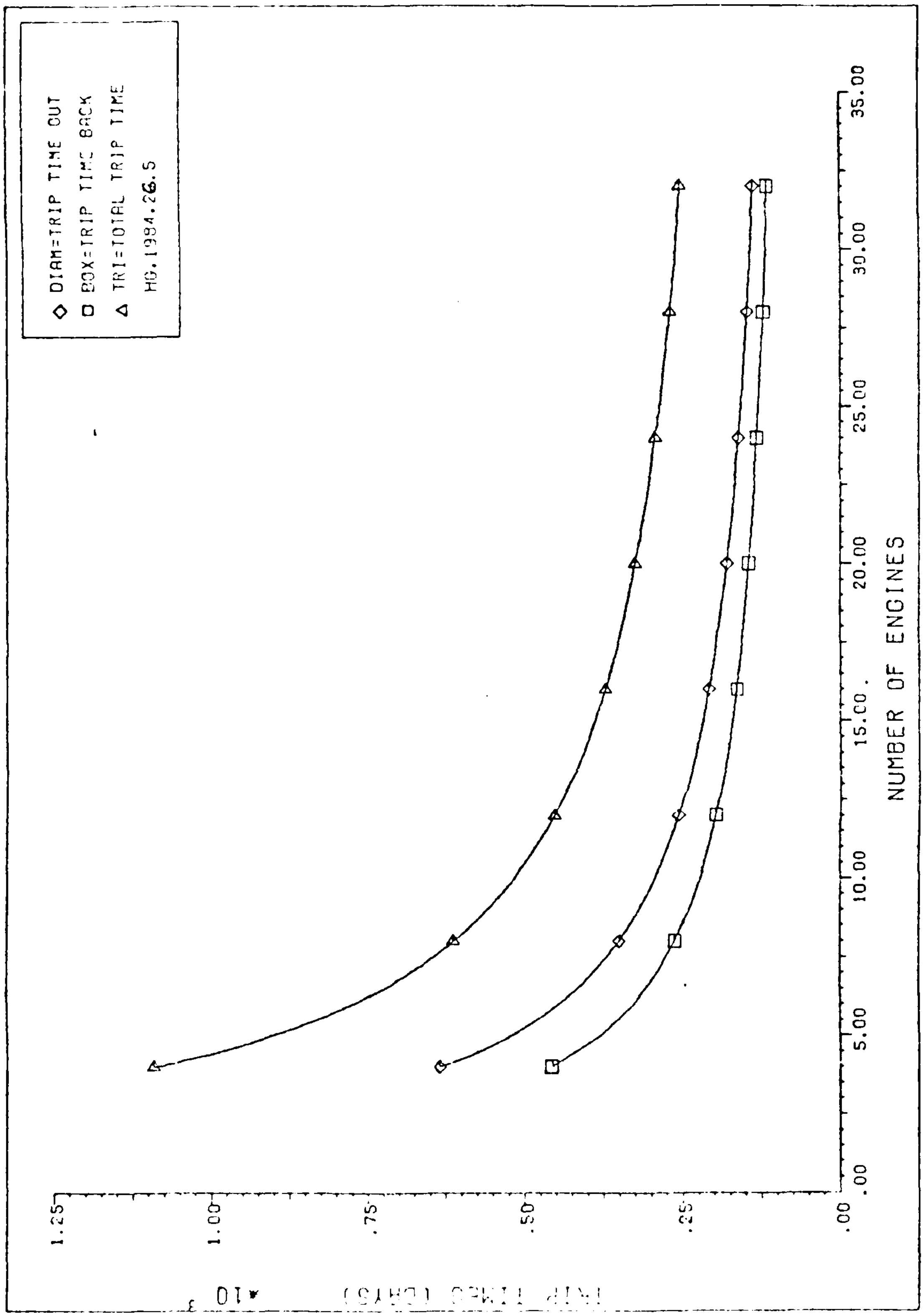
TOTAL COST VS. NUMBER OF ENGINES. HG,AR,XE 1984

Figure 4-1.



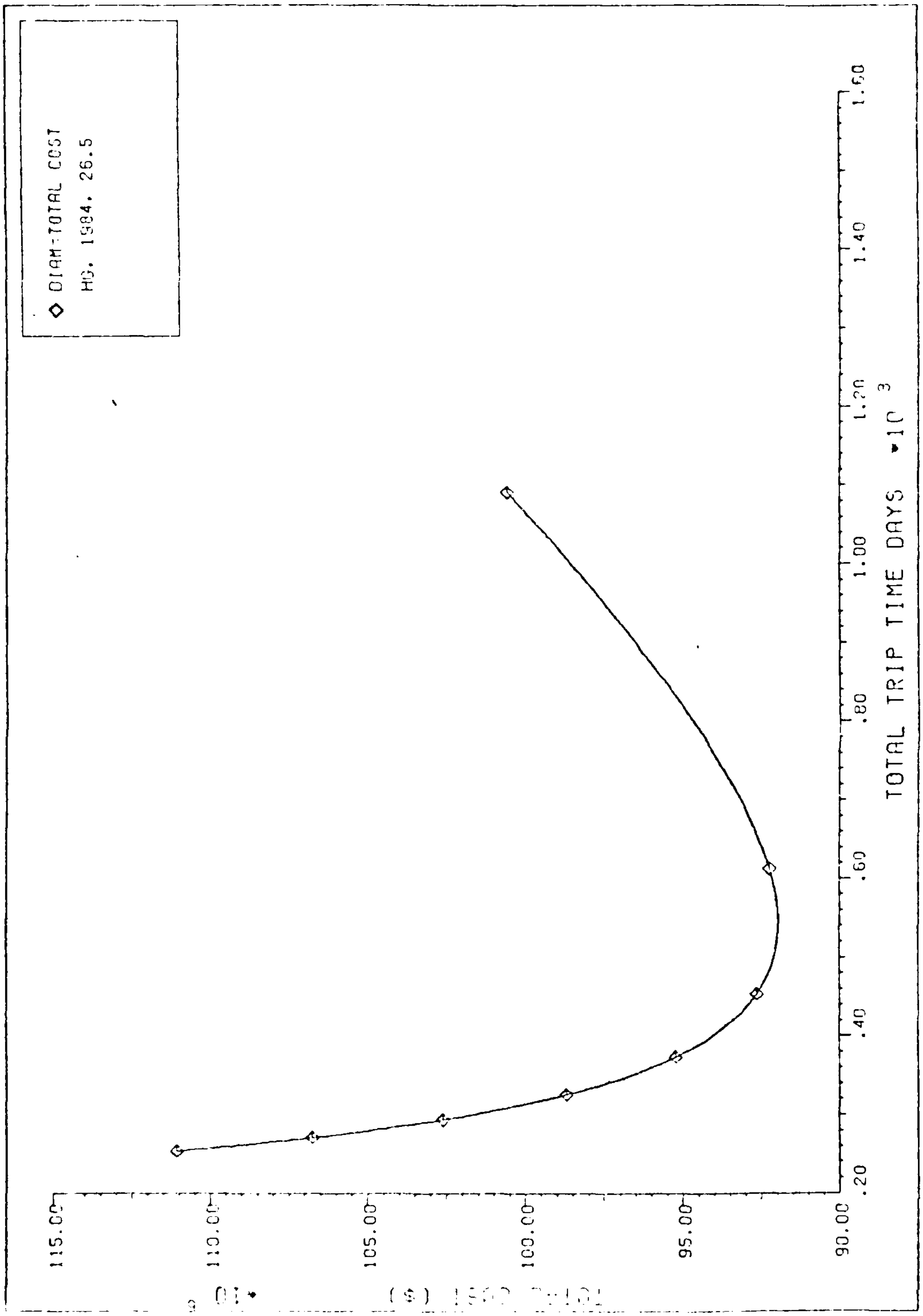
COSTS VS. NUMBER OF ENGINES HG.1984.26.5

Figure 4-2.



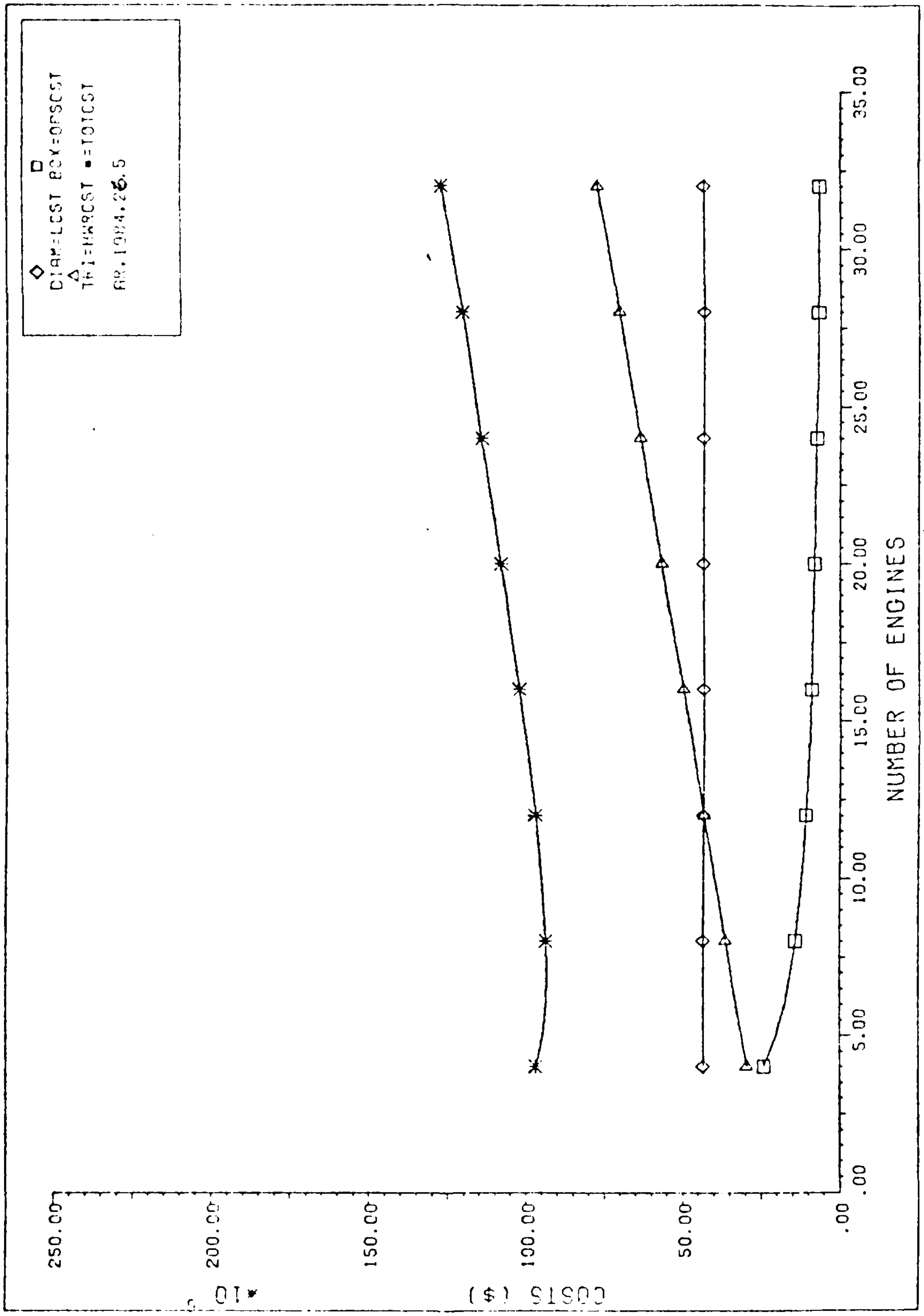
TRIP TIMES VS. NUMBER OF ENGINES HG.1984.26.5

Figure 4-3.



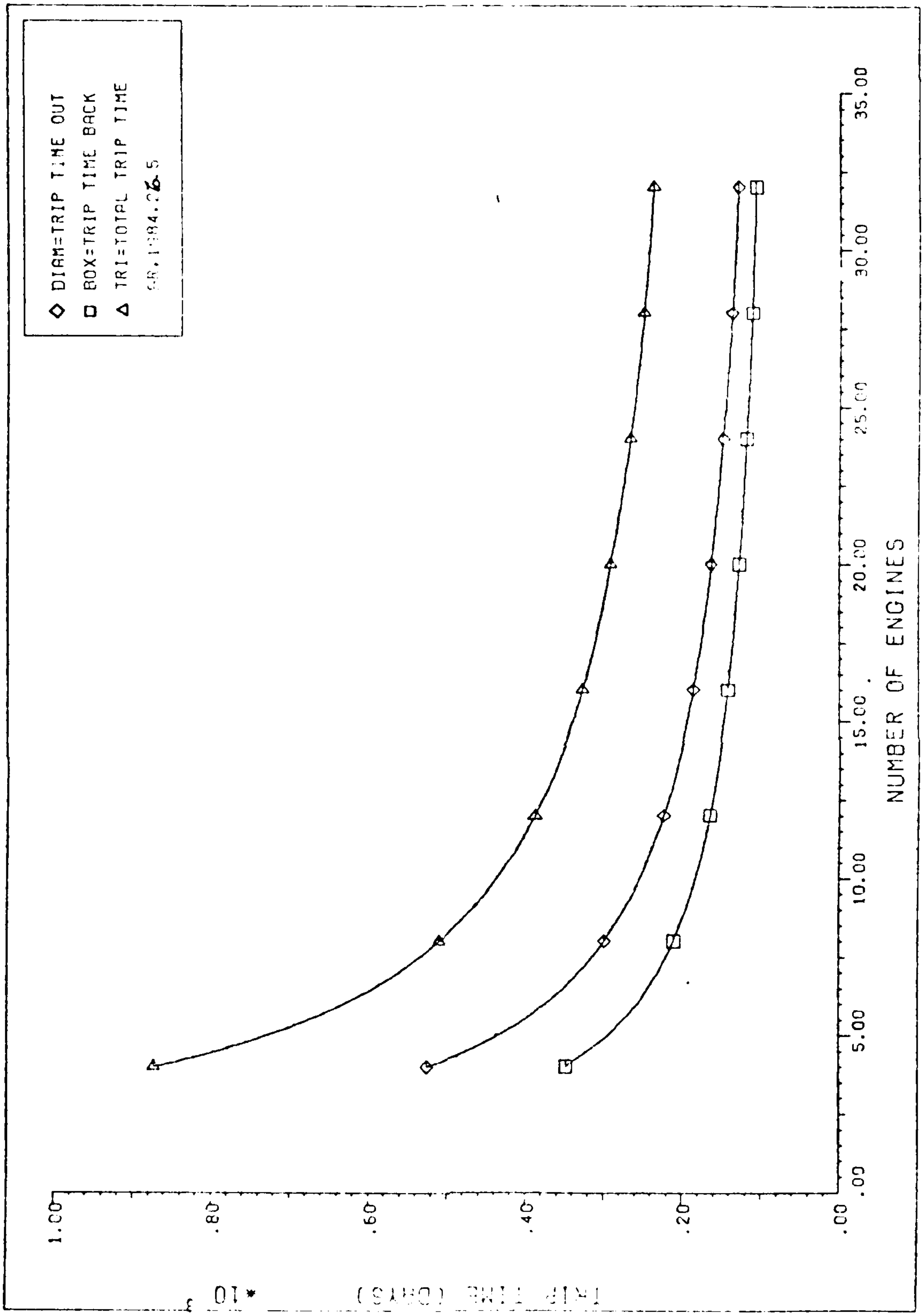
TOTAL COST VS. TRIP TIME HG. 1984. 26.5

Figure 4-4.



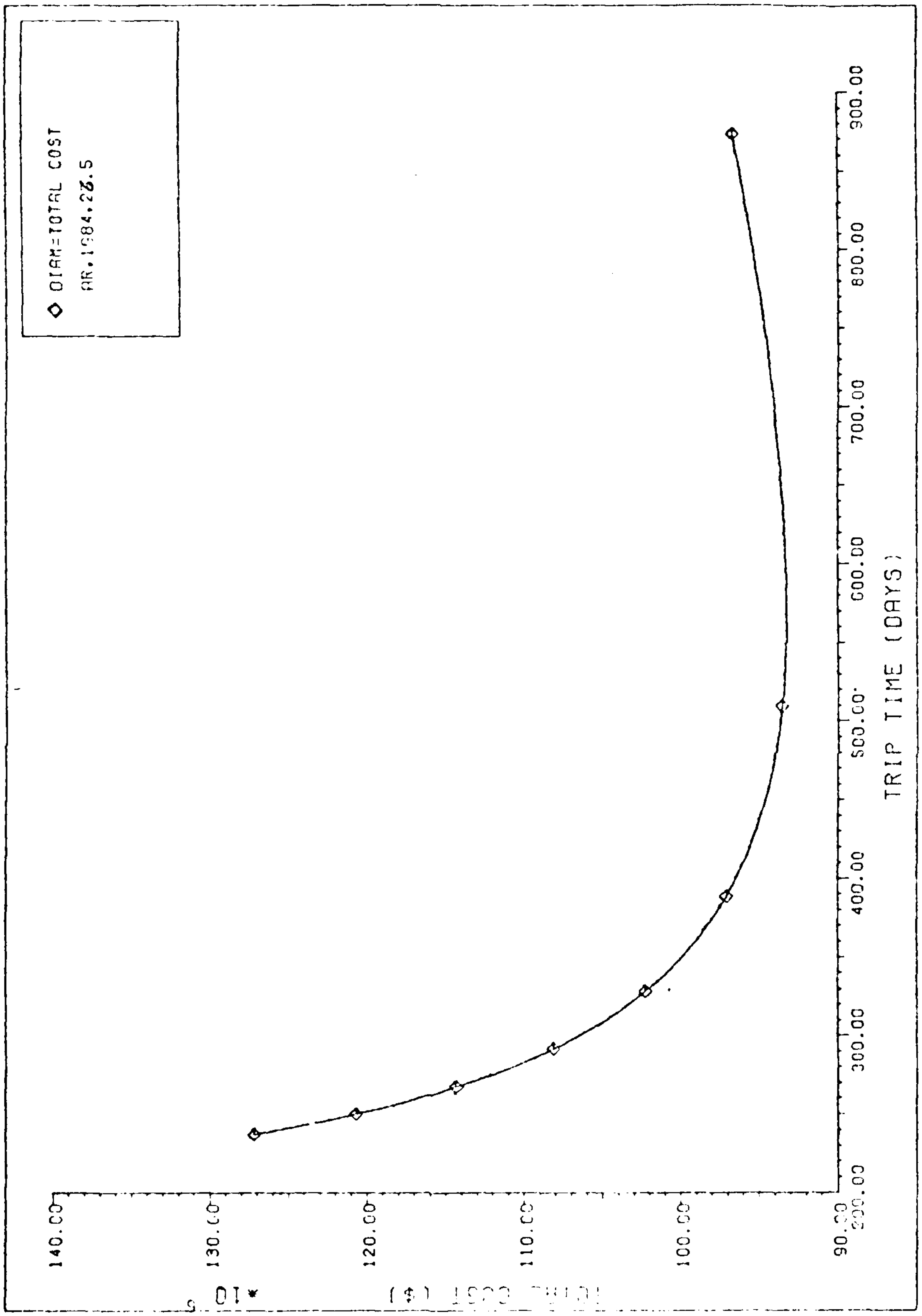
COSTS VS. NUMBER OF ENGINES AR.1984.26.5

Figure 4-5.



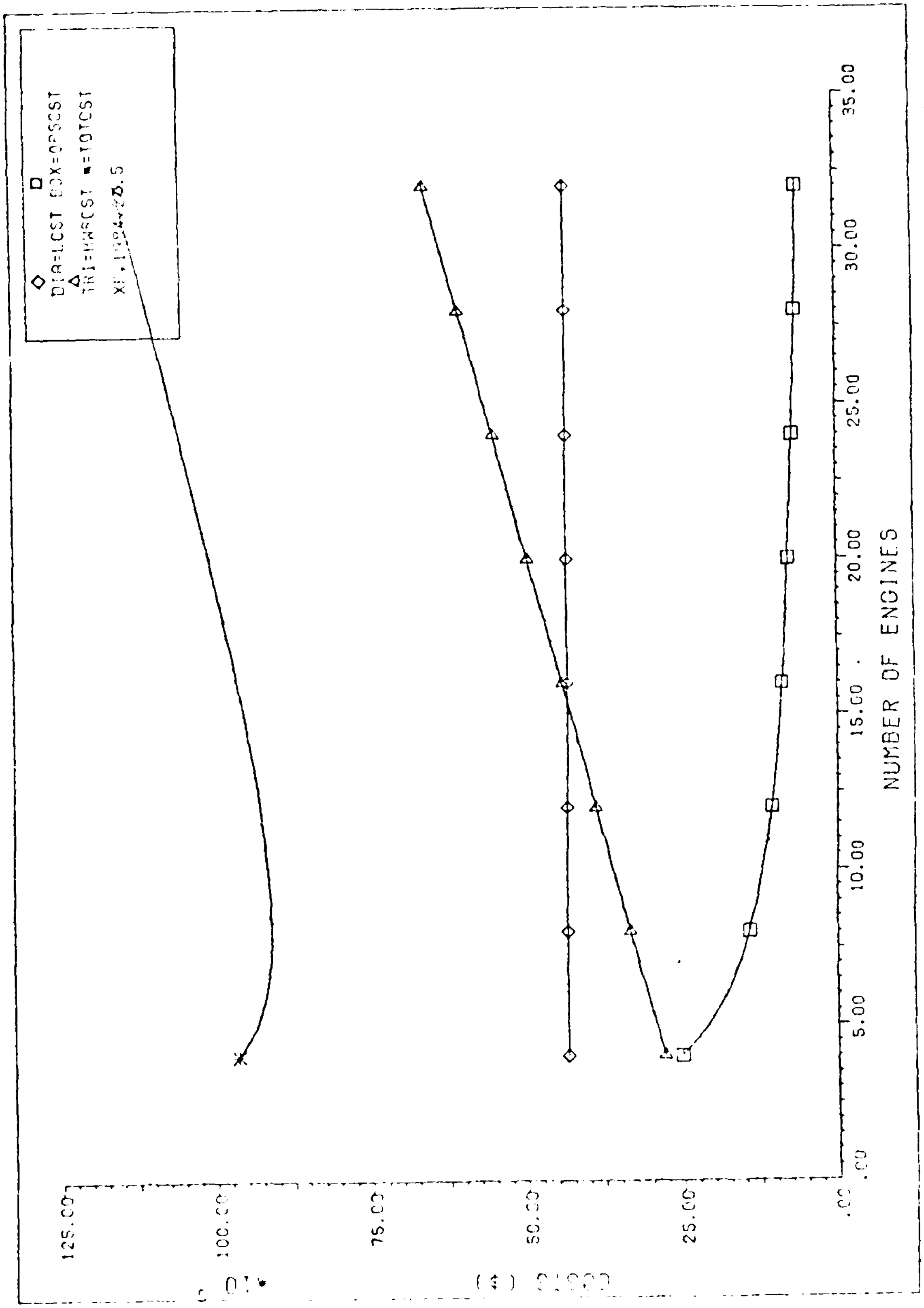
TRIP TIME VS. NUMBER OF ENGINES AR.1984.26.5

Figure 4-6.



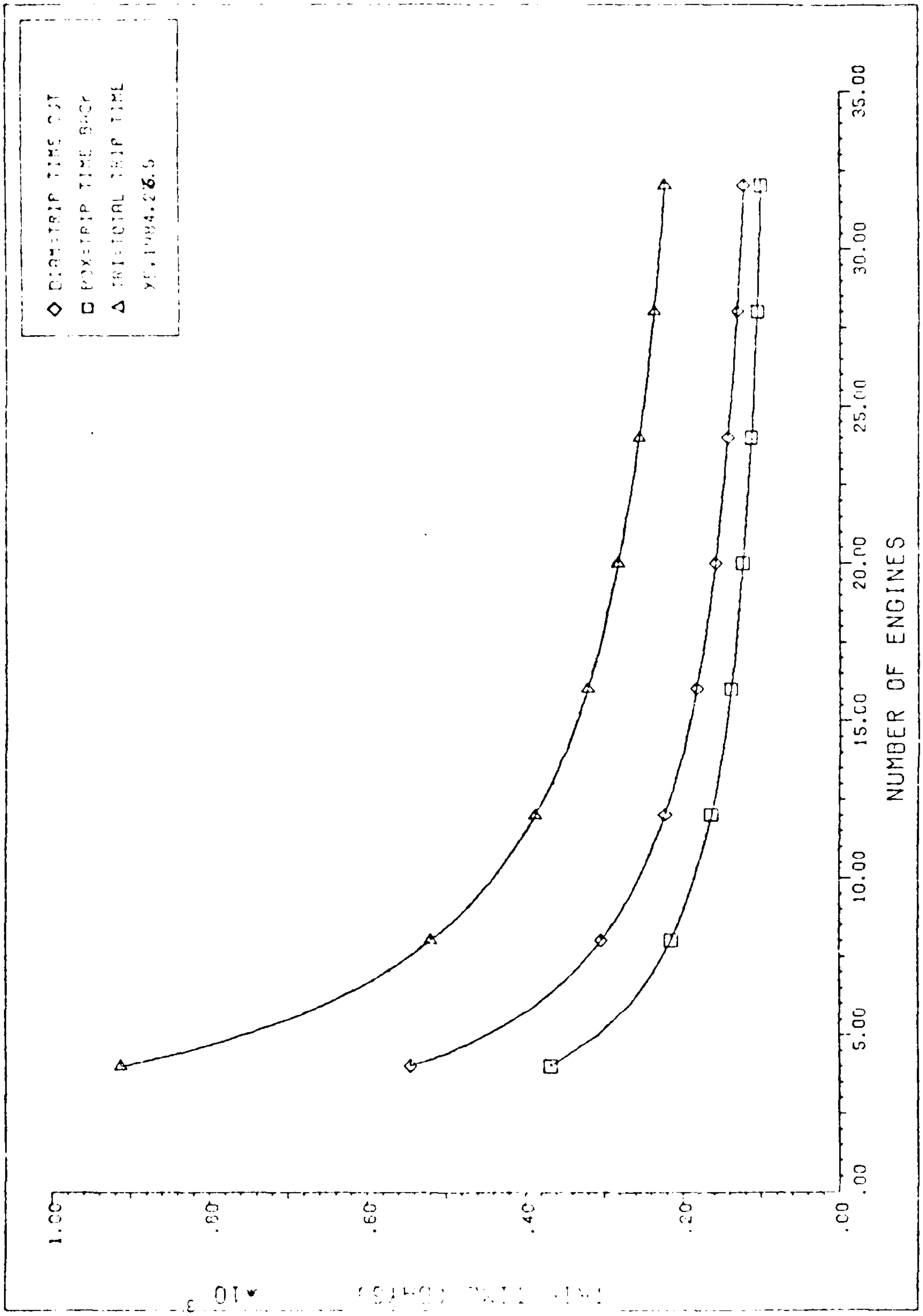
TOTAL COST VS. TRIP TIME AR.1984.26.5

Figure 4-7.



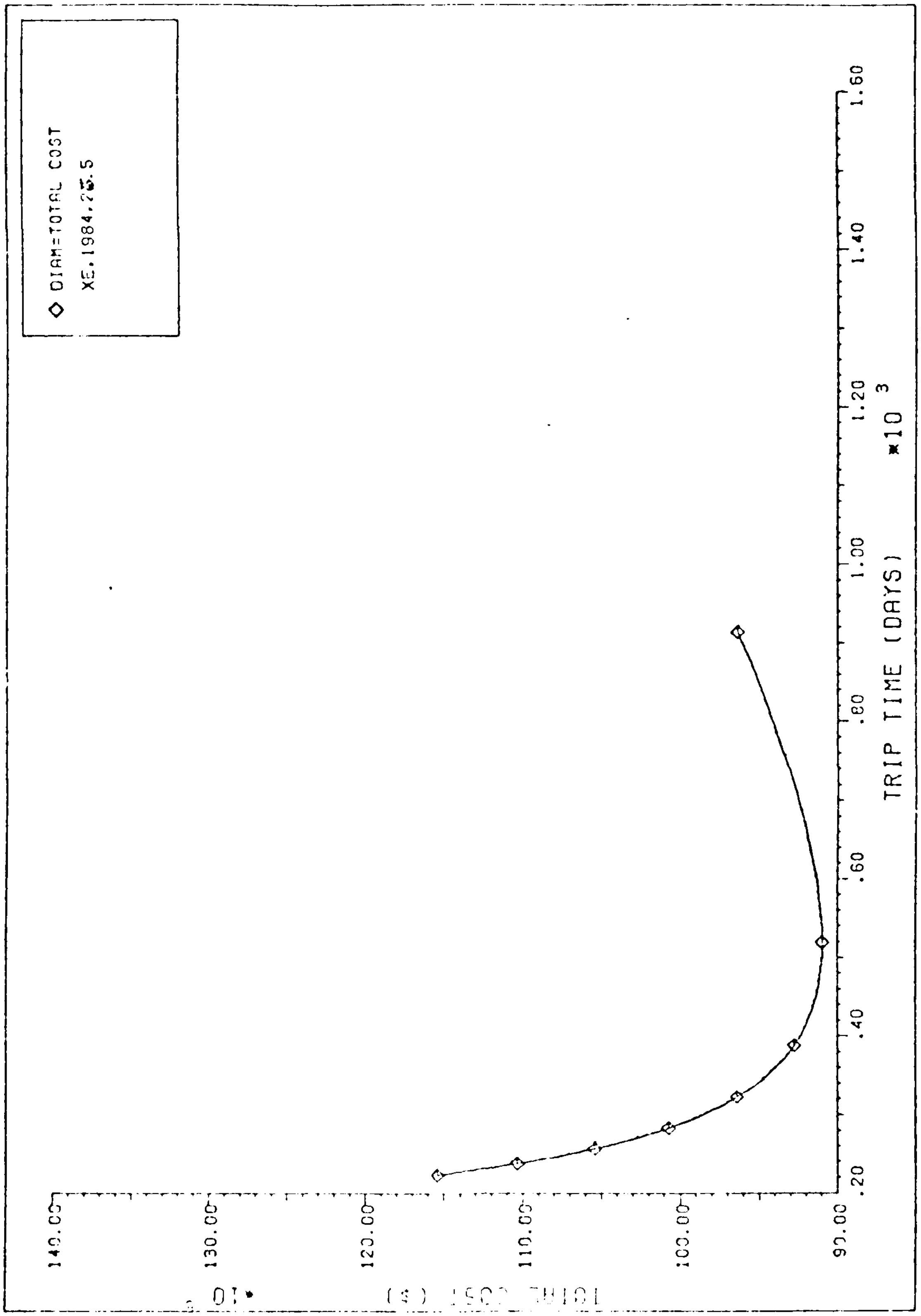
COSTS VS. NUMBER OF ENGINE XB.1984.26.5

Figure 4-8.



TRIP TIME VS. NUMBER OF ENGINES YR.1984.26.5

Figure 4-9.



TOTAL COST VS. TRIP TIME XE.1984.26.5

Figure 4-10.

with the other individual masses of the systems, results in a large difference in the total system masses (6074, 4908, and 5144 Kg). System mass directly influences the trip time, and hence cost. According to some studies (40, 47), system mass is the most crucial factor affecting an electric OTV system.

The total cost graph (Figure 4-1) shows that the curves intersect each other. As the number of engines increases, the fuel source for the minimum cost system changes from Argon to Mercury. This results from the different fuel source engines having different power requirements, masses, and specific impulse values. These all combine to produce this switch. In particular, it can be seen from the graphs that the operations costs become closer, while there is a constantly increasing separation in the hardware costs. The launch costs remain the same for each system because this cost is based on the size of the system, and all of the systems are the same length (see Appendix C). Because the trip times are unacceptable in all cases, we will now consider the expected improvements to be made by the 1995 time frame.

The table below represents improvements which are expected and predicted to occur by the 1995 time frame.

	Hg	Ar	Xe
Isp (sec)	3000	6000	3500
Power/Thrust (Kw/N)	17.7	35	20.7
Thrust system efficiency	.83	.84	.83
Thrust system mass/Thrust (Kg/N)	196	350	207

These improvements are derived from expected improvements as predicted by several sources (15, 64, 66). The cost per engine system is expected to be reduced by 50%, mainly as a function of improvements in the power processor area of the engine system (66). These improvements give the table below which represents the engine parameters used for the 1995 runs.

	Hg	Ar	Xe
Thruster efficiency (Nsubt)	.9222	.933	.92
Isp (sec)	3000	6000	3500
Power required (Kw)	2.283	4.515	2.67
Engine system mass (kg)	40	45	40
Cost /Engine system ($/Kg)	8,750	8,750	8,750

The table below represents the results of the 1995 runs with delta inclination still at 26.5 degrees.

1995 Delinc = 26.5

	Hg	Ar	Xe
Total cost (millions $)	86.02416	88.29218	86.02538
Total trip time (days)	420.65	483.82	416.32
Number of engines at optimum point	12	8	11

Now Mercury has a very slight lead in terms of money, but still has not gone ahead of Xenon in terms of trip time. This differs from the 1984 runs where the Xenon system had the minimum cost. In terms of total systems however, these two may be said to be relatively equal at this point. The trip time, however, is still unacceptable to the user. Because of this, the results of the computer runs were presented to the user, and he decided that he would be willing to trade a

higher cost for a shorter trip time. The user chose the maximum number of allowable engines (restricted by 100 Kw power source) as his operating point based on his trip time requirements and accepted the extra costs incurred. In reality, this may prove to be the only other possible operating point, as the costs of a variable design system may be large. The table below gives the options considered for his choice, the mercury system at 43 engines, and the xenon system at 37 engines.

Hg, 1995, 43 engines, 26.5 delinc

Launch cost (millions $)	43.333
Operations cost	5.236
Hardware cost	53.412
Total cost	101.9822
Trip time out (days)	103.53
Total trip time	190.59

Xe, 1995, 37 engines, 26.5 delinc

Launch cost (millions $)	43.333
Operations cost	5.333
Hardware cost	51.944
Total cost	100.6112
Trip time out (days)	106.64
Total trip time	194.65

Because the trip time out slightly exceeded the users requested time of 90 days, another possibility was considered. The shuttle has the capability and is planned to deliver payloads directly into a 55 degree orbit inclination (delinc=0). If this option is used and the flight is not a dedicated mission (where all costs are assumed by the single,

dedicated user), the trip time changes dramatically while the launch costs remain the same. The program was run for this option and the cost optimum points were again very close. However, the trip time remained unacceptable to the user. Therefore, 43 engines were again chosen for this option for mercury and 37 engines for xenon for the same reasons as given in the first case. The results are indicated below.

Hg, 1995, 43 engines, 0 delinc

Launch cost (millions $)	43.333
Operations cost	3.376
Hardware cost	53.395
Total cost	100.105
Trip time out (days)	67.56
Total trip time	122.89

Xe, 1995, 37 engines, 0 delinc

Launch cost (millions $)	43.333
Operations cost	3.554
Hardware cost	51.932
Total cost	98.820
Trip time out (days)	71.86
Total trip time	129.38

Because the difference between Mercury and Xenon is very small, Xenon will be compared against the solar powered and chemical OTV's for the cost analysis. Although its trip time out is 4 days greater, Xenon was chosen because its total cost is marginally lower. A further consideration for using Xenon is the environmental impacts of using Mercury (20). While not directly considered here, it could become a strong deciding factor if the use of large amounts of mercury is

necessary.

In considering why the optimum fuel source changed between the 1984 and 1995 initial runs, a simple attempt at normalizing the engine specific impulse over the other engine parameters was tried. When the Isp was normalized over thrust, mass of engines, and power, the following results were obtained.

Isp/thrust-mass-power (sec/N-kg-Kw)

1984			1995		
Hg	Ar	Xe	Hg	Ar	Xe
144.05	152.76	182.40	254.6628	228.92	254.042

We can see from the above numbers that the results show a definite relation to these figures. In the 1984 case, Xenon is by far the winner, giving the best trip time and the minimum cost. It also has, by far, the highest normalized specific impulse. In the 1995 case, Mercury and Xenon are almost equal in terms of both cost and trip time. Again, the normalized specific impulse follows this pattern, being almost equal in the two cases.

Before running the best case against the alternative chemical OTV's, there are two more considerations to be addressed. The first is that the normal shuttle deployment altitude is 300 Kilometers. Though this makes a negligable difference in the total cost, the system was run for this initial orbit altitude. The results are:

1995, 0 delinc, initial radius = 300 Km

	Hg	Xe
Launch cost (millions $)	43.33	43.33
Operations cost	3.30	3.48
Hardware cost	53.39	51.93
Total cost	100.03	98.75
Trip time out (days)	66.21	70.45
Total trip time	120.37	126.89

We can also assume that the cost of the reactor may not be the maximum 40 million but the minimum 20 million dollars (17). This reduces the results by 20 million dollars in all cases and changes the minimum points, but the final fuel choice remains the same. These are the final cost figures which will be used for comparison with the chemical systems. The minimum reactor cost is used here because if the decision is made to use electric OTV's, this will be the representative cost of the reactors (17).

Minimum reactor cost
1995, 0 delinc, initial radius = 300 Km

	Hg	Xe
Launch cost (millions $)	43.33	43.33
Operations cost	3.30	3.48
Hardware cost	34.81	33.02
Total cost	81.45	79.84
Trip time out (days)	66.21	70.45
Total trip time	120.37	126.89

CHAPTER V. SOLAR POWER ANALYSIS

METHODOLOGY

The basic methodology consisted of using the system cost model developed in the Boeing study (26). The costs of an EOTV system are determined using various combinations of propellant and solar array types. The overall goal was to find the engine propellant /solar array combination which produced the lowest total mission cost while meeting the user imposed constraint of an outbound triptime of 90 days or less. If the engine propellant/solar array combination which produced the lowest total mission cost did not satisfy the time constraint then an operating point was found by picking a point on the total cost / mission time tradeoff curve at which the user felt both the cost and the time were still acceptable. The EOTV costs were then compared to the costs for the present options for upper stages - PAM DII, IUS, and CENTAUR-G. Thus it was determined whether an EOTV is a cost effective alternative for the deployment of the GPS Block 3 satellite.

The system cost model that was used will be presented first. While most of it comes from the Boeing study (26), portions were modified as needed and some portions such as the transfer time equation and the velocity change equation were obtained from the R. M. Jones article (40). This author will not derive the equations, especially those having to do with the orbit transfer. Derivations for these equations can

be partially found in the source articles.

Next, the method used to derive the values for the solar array degradation will be briefly explained. The derivation of values for the solar array specific mass and specific power can be found in Appendix G.

SYSTEM COST MODEL

The calculation process used is illustrated below:

```
PAYLOAD CHARACTERISTICS --------->  ELECTRIC

TRAJECTORY REQUIREMENTS --------->  PROPULSION          MISSION
                                                 ---->
POWER SOURCE CHARACTERISTICS --->  SYSTEM               COSTS

EARTH LAUNCH SYSTEM ------------>  MODEL
CHARACTERISTICS
```

Figure 5-1. SOLAR POWER ANALYSIS CALCULATION PROCESS

The mission cost equation is as follows:

$$CM = CEPS + CSA + CETO + CTT + CP \qquad (5\text{-}1)$$

CM = total mission cost from the earth's surface to the final orbit

CEPS = purchase cost of the electric propulsion system

CSA = purchase cost of the solar array

CETO = cost of the launch to LEO

CTT = cost penalty due to non-negligible transfer time

CP = purchase cost of the propellant

One measure of a system could be CM or it could be the cost factor (CSTFAC) which is measured in $/kg of payload delivered.

$$CSTFAC = CM / MPL \tag{5-2}$$

MPL = mass of the payload (kg)

Most of the above costs are easily derived. The cost of the electric propulsion system which includes the electric thrusters, the power processing units (PPUs), the support structure, the propellant tanks and lines, and the radiators, is based on a per unit cost for the engine system. A constant cost for a guidance and control unit is also included.

$$CEPS = (NENG \times GEPS) + CMAV \tag{5-3}$$

NENG = number of engines

GEPS = unit cost of an engine system ($/engine)

CMAV = cost of a guidance and control unit ($)

Similarly, the mass of the EPS is:

$$MEPS = (NENG \times MENG) + MAV \tag{5-4}$$

MEPS = mass of the EPS (kg)

MENG = mass of an engine system (kg/engine)

MAV = mass of the guidance and control unit (kg)

The cost of the solar array (CSA) and the mass of the solar array (MSA) are found by:

$$CSA = GSA \times PNOM \tag{5-5}$$

GSA = specific cost of the solar array ($/KW)

PNOM = nominal power (KW)

$$MSA = ASA \times PNOM \tag{5-6}$$

ASA = specific mass of the solar array (kg/KW)

$$PNOM = PREQ / (1 - R) \tag{5-7}$$

$$PREQ = NENG \times PENG \tag{5-8}$$

PREQ = required power (KW)

PENG = input power required at the PPU (KW/engine)

R = degradation factor for the solar array

The cost of the launch to LEO is found by using the shuttle launch cost equation. The two forms of this equation are:

$$CETO = \frac{(MT / 29484)}{.75} \times 65 \tag{5-9}$$

$$CETO = \frac{(LT / 60)}{.75} \times 65 \tag{5-10}$$

MT = total mass of the shuttle payload including shuttle adaptor hardware (kg)

29484 = maximun shuttle payload (kg) for a given LEO. This value is for the nominal orbit of 28.5 degrees inclination and 160 nautical miles. For an orbit of 55 degrees inclination, this would be 25855 kg.

65 = FY84 cost ($ Million)

LT = total payload length (ft)

60 = shuttle payload bay length

The higher of the two values for CETO is used. This corresponds to the higher of the weight factor or the length factor of the payload. After initial computations were made,

it was determined that the EOTV using solar arrays would have a greater weight factor than length factor. Thus only the first equation is used in the final program.

CTT, the cost penalty resulting from non-negligible transfer time is composed solely of the cost to track and control the satellite during the orbit transfer.

$$CTT = GOPS \times T \qquad (5\text{-}11)$$

GOPS = satellite control operations cost ($/year)

T = transfer time (seconds)

The transfer time (T) is computed as follows:

$$T = \frac{MP\ (g \times ISP)^2 \times (1 + PHI \times (1 + TD))}{2 \times N \times PREQ} \qquad (5\text{-}12)$$

MP = mass of the propellant (kg)

g = 9.8 m/sec2

ISP = specific impulse (seconds)

PHI = penalty to account for time in the earth's shadow

TD = time penalty for engine restart

N = system efficiency

The propellant mass (MP) is calculated from:

$$MP = \frac{(MPL + MEPS + MSA + MPR) \times (1 - e^{(-X)})}{e^{(-X)}} \qquad (5\text{-}13)$$

where

$$X = \frac{DELV \times (1 + D)}{.0098 \times ISP} \tag{5-14}$$

$$DELV = \left[VFIN^2 + VINT^2 - 2 \times VFIN \times VINT \times COS\left(\frac{\pi\ DELI}{2}\right) \right]^{1/2} \tag{5-15}$$

DELI = change in inclination (degrees)

VFIN = velocity in final orbit (km/sec)

VINT = velocity in initial orbit (km/sec)

DELV = change in velocity (km/sec)

D = drag penalty factor

MPR = mass of propellant for the return trip (kg)

MPR is calculated using the same equation as MP except that the first term is *made up only* of MEPS and MSA since that is all that returns to LEO.

The propellant cost is found by:

$$CP = MP \times GP \tag{5-16}$$

GP = specific cost of propellant ($/kg)

The parameters that are input into the above cost model are summarized below:

For the solar array:

R = degradation factor

ASA = solar array specific mass (kg/KW)

GSA = solar array specific cost ($/KW)

For the EPS:

NENG = number of engines

ISP = specific impulse (seconds)

PENG = input power to the PPU per engine (KW/engine)

MENG = mass of the engine system (kg/engine)

GEPS = unit cost of an engine system ($/engine)

N = system efficiency

For the payload:

MPL = mass of the payload (kg)

For the trajectory:

HINT = initial orbit altitude (km)

HFIN = final orbit altitude (km)

IINT = initial orbit inclination (degrees)

IFIN = final orbit inclination (degrees)

The values for the input parameters will come from the data base gathered from the literature review.

SOLAR ARRAY DEGRADATION

A major challenge in the thesis was to formulate a methodology to derive a value for the degradation of the solar arrays as a result of passage thru the Van Allen belts.

Values for the fluence levels in equivalent 1 MeV electrons/sq cm are available in the Solar Cell Radiation Handbook (4). These are presented in tables for every 10 degrees of inclination with data for altitudes from 150 to 19327 nautical miles and for array shield thicknesses of 0, 1, 3, 6, 12, 20, 30, and 60 mils (1 mil = 1/1000 inch). Thus

the problem of calculating fluence levels would be solved if the position of the EOTV could be determined at any given time.

Using Captain Alfano's thesis (3:7) for the transfer between two circular orbits, the change in the semimajor axis (a) can be written as:

$$\frac{da}{dt} = \frac{2\,V}{\sqrt{\frac{\mu}{a^3}}} \qquad (5\text{-}17)$$

where V = tangential acceleration component

μ = gravitational parameter = 3.986×10^5 km^3/sec^2

a = semimajor axis (radius for a circular orbit)

Integrating with respect to time from 0 to 86400 seconds (1 day) will yield the change in semimajor axis in one day. Because of the small thrust levels, Δa will be very small compared to the semimajor axis. Thus for one day, the semimajor axis is considered constant. This yields:

$$\Delta a = 2\,V \left[\frac{a^3}{\mu}\right]^{1/2} (86400) \qquad (5\text{-}18)$$

For the EOTVs being examined, the tangential acceleration component (V) is equal to the thrust divided by the mass of the EOTV (MT). Thrust is equal to the number of engines (NENG) multiplied by the thrust per engine (129 mN). Thus:

$$V = \frac{NENG \times (.129)}{MT} \qquad (5\text{-}19)$$

Using these equations it was possible to write a program that calculated Δa for one day, updated the value of a, calculated Δa for the next day, and iterated until the desired semimajor axis value was reached.

To determine the inclination of the transfer orbit, the "Universal Chart for Orbit Transfer" developed in Captain Alfano's thesis was used (3:37). This chart which represents the time optimum transfer orbit for low acceleration systems, plots the inclination change against the semimajor axis. Thus the transfer orbit inclination corresponding to any given value of semimajor axis was read directly from the chart.

Using these approximations for the EOTV position in the transfer orbit, it was possible to go into the fluence tables in the Solar Cell Radiation Handbook and calculate the fluence for a given transfer orbit and an array shield thickness. Here again, a simple program was written to interpolate between the values given in the book and then to calculate a cumulative fluence level.

Once the fluence levels were calculated, it was possible to determine the normalized maximum power (NPmax) level for any given silicon cell type by simply looking up the plots of NPmax vs 1 MeV electron fluence contained in the same handbook. The value for degradation (R) to be used in the main program is simply:

$$R = 1 - NPmax \tag{5-20}$$

It is nearly impossible to accurately calculate the exact fluence levels for a satellite transiting the Van Allen belts mainly because the proton and electron populations in these belts are not constant. Therefore, predictions are made based on historical averages. The method developed by the author establishes the representative position of the EOTV, not the exact position. However, since the overall goal is only to establish first order estimates of the radiation damage to the solar cells, this orbit position determination method and the use of historical averages for radiation levels are appropriate.

CHAPTER VI. SOLAR RESULTS AND ANALYSIS

CALCULATING THE FLUENCE

Initial computer runs of the system model used input parameter values found in the available literature. The triptimes for the EOTV were found to be approximately 150 days for a transfer from 28.5 to 55 degrees and 110 days if no inclination change is performed.

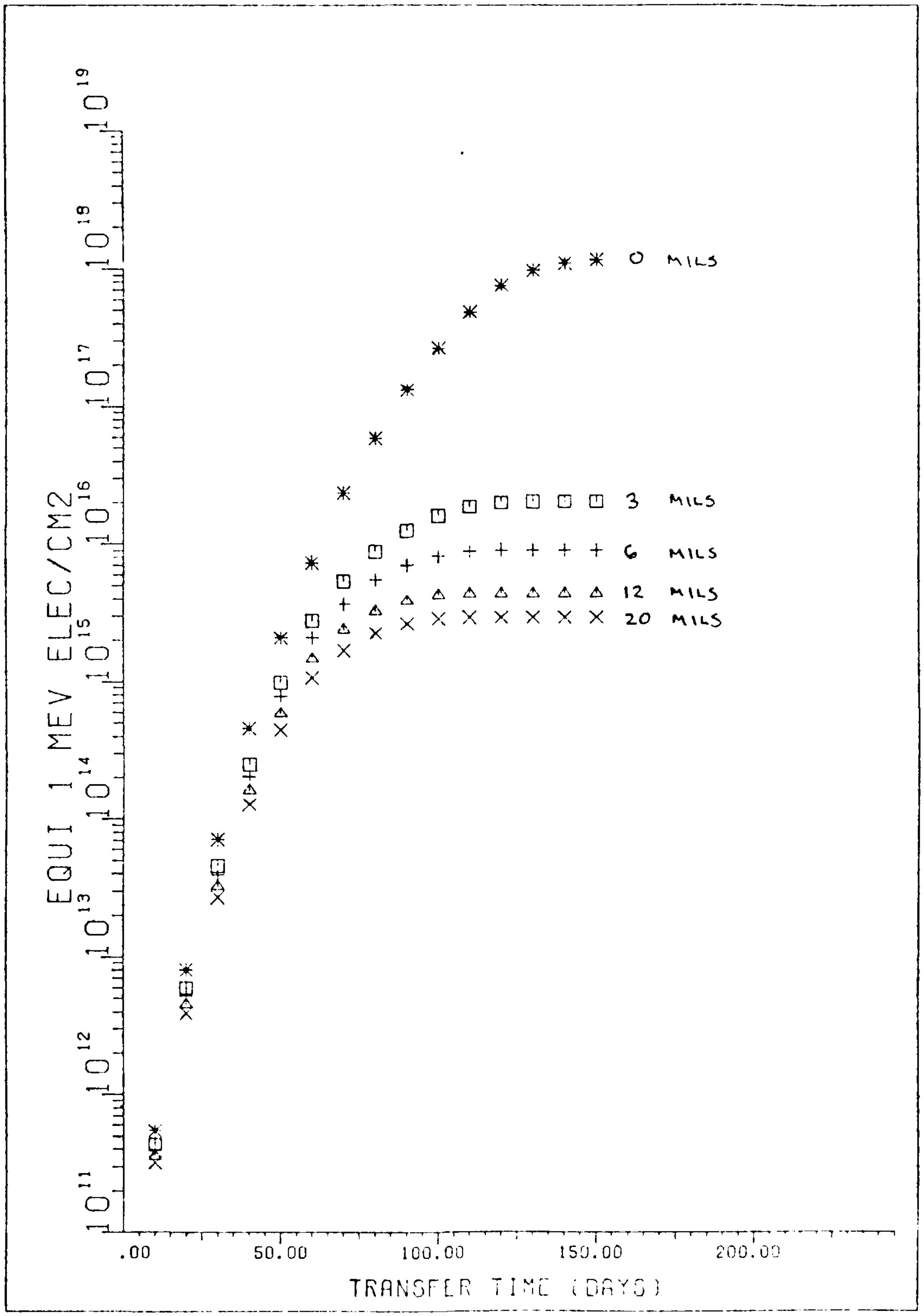
Using the method presented earlier, the fluence levels were calculated for arrays with covers of 0, .003, .006, .012, and .02 inches of microsheet. Because protons in the Van Allen belts travel in all directions, damage to the silicon cells is caused not only by protons entering thru the top (or front) of the array but also by those entering thru the bottom (or back) of the array. Thus the fluence levels must be adjusted to account for the protons entering thru the back of the array. The backing of the solar array blankets that were considered in this thesis consists of two layers of 5 mil kapton. The portion of the silicon cell that is subject to the degradation, the N/P junction, is located at the top of the cell. Therefore, from the back, it is shielded by 10 mils of kapton and almost all of the 4 mils of silicon. This is equivalent to approximately 10 mils of microsheet cover. The fluence thru this amount of back cover was added to the fluence calculated for the given front covers. These adjusted values were then used to calculate the degradation for a 4 mil, 10 ohm-cm, N/P silicon cell.

The results of these calculations are found in Figures 6-1 - 6-4. Only the effects of the protons were considered because their fluence levels were generally two orders of magnitude greater than those for electrons and they are therefore the predominant damage inflictors.

Referring to Figure 6-1, note that the fluence curves level off. This occurs because the majority of the high energy protons are at the lower levels of the Van Allen belts. The peak fluences occur from 5000 to 10000 km depending on inclination. Once the EOTV passes this altitude, the fluence levels drop off considerably and thus the cumulative fluence levels off.

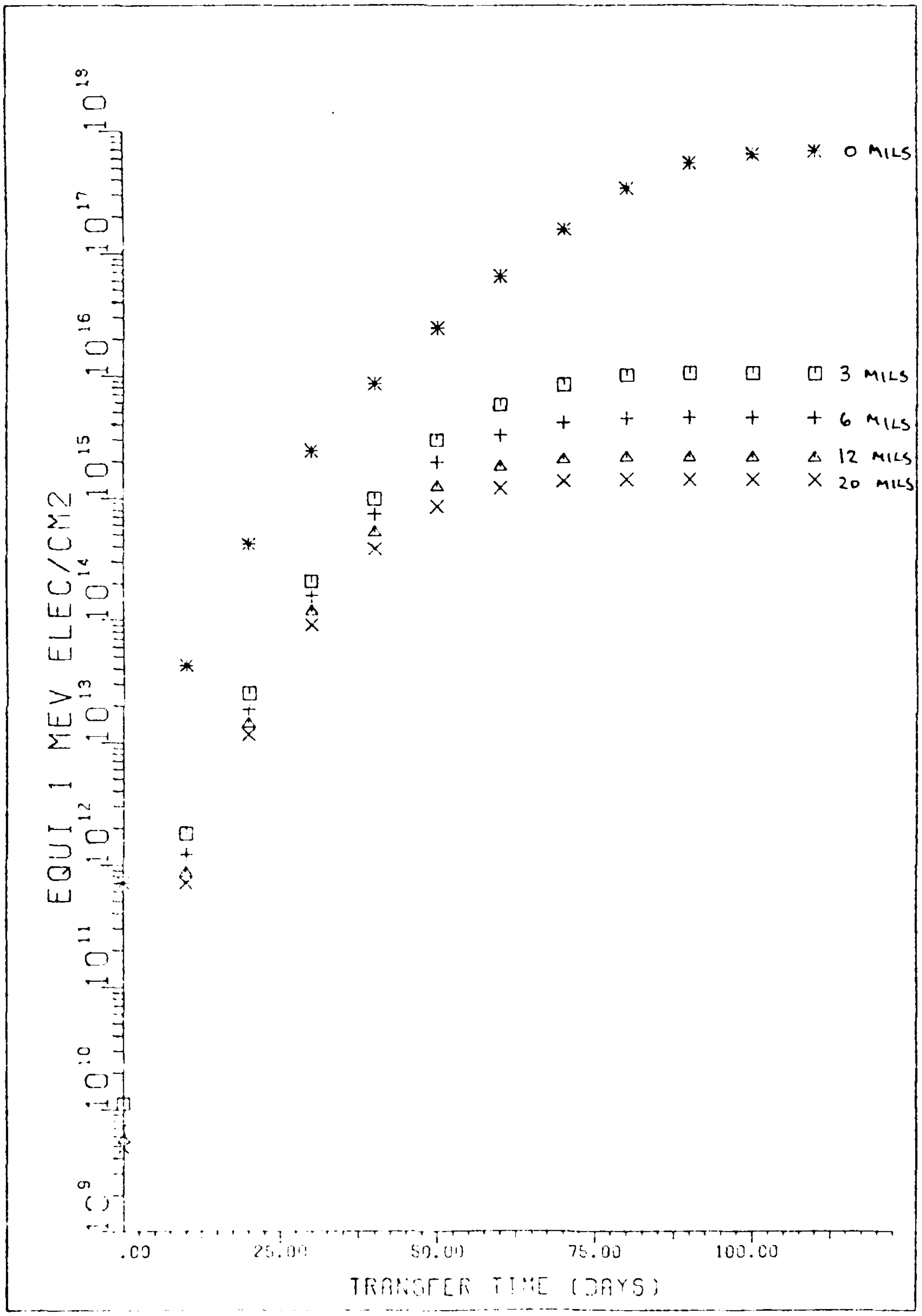
The graphs of the solar array degradation for the 110 day transfer with no inclination change, Figure 6-4, show that the solar arrays degrade faster than those in a 150 day transfer. This is due to the EOTV reaching the heart of the Van Allen belts quicker when no inclination change is needed. This causes the solar array degradation to take place earlier in the transfer. The faster transfer also means the EOTV penetrates the Van Allen belts in less time. The resulting lower fluence produces a lower overall degradation which the graphs also show.

After subsequent runs of the system model using updated values for the input parameters, the triptimes for the EOTV were now in the neighborhood of 70 days. Therefore, the fluence levels and degradation values for the transfer from 28.5 to 55 degrees and the transfer at 55 degrees were



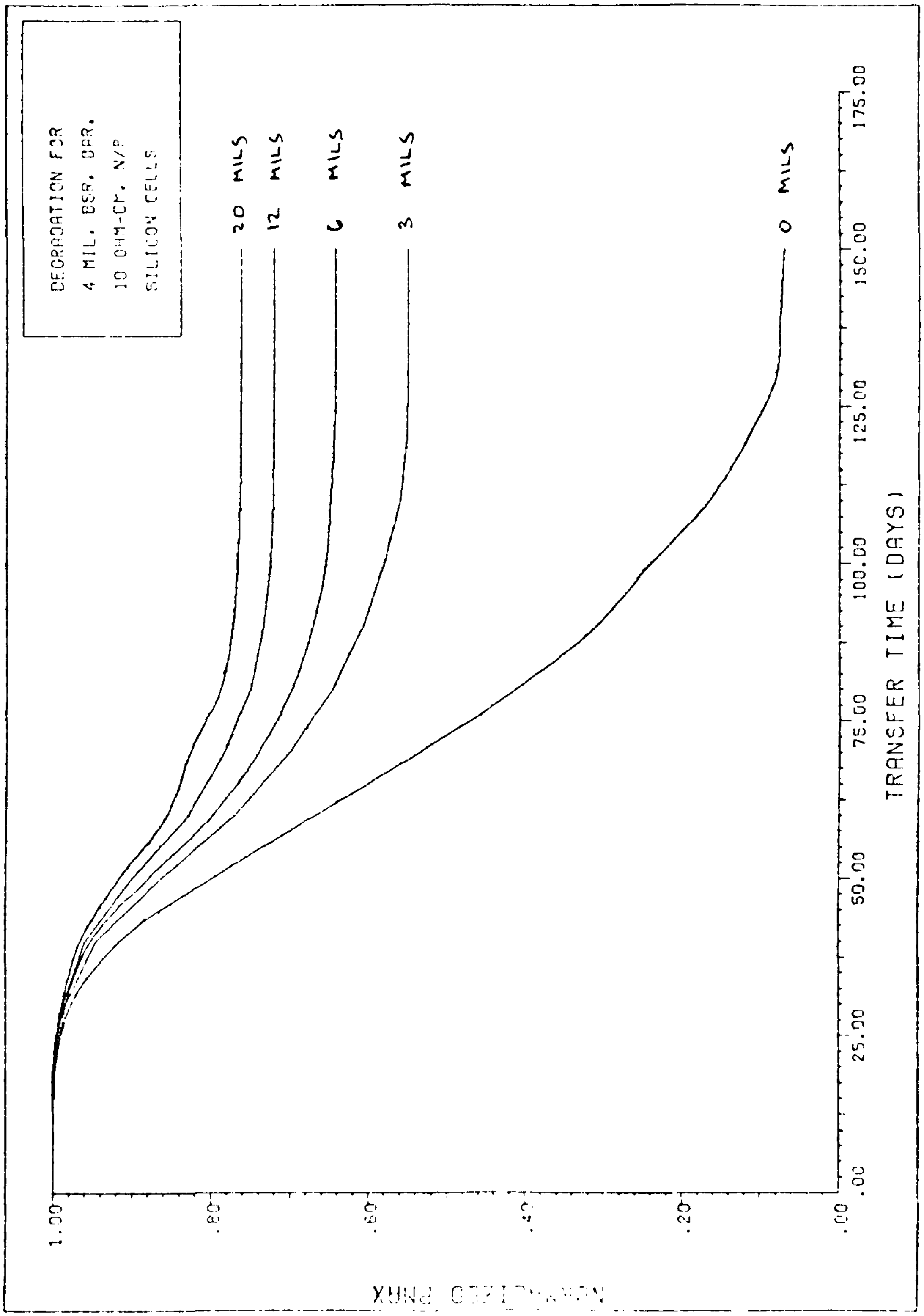
PROTON FLUENCE FOR 150 DAY TRANSFER: 28.5° TO 55°

Figure 6-1

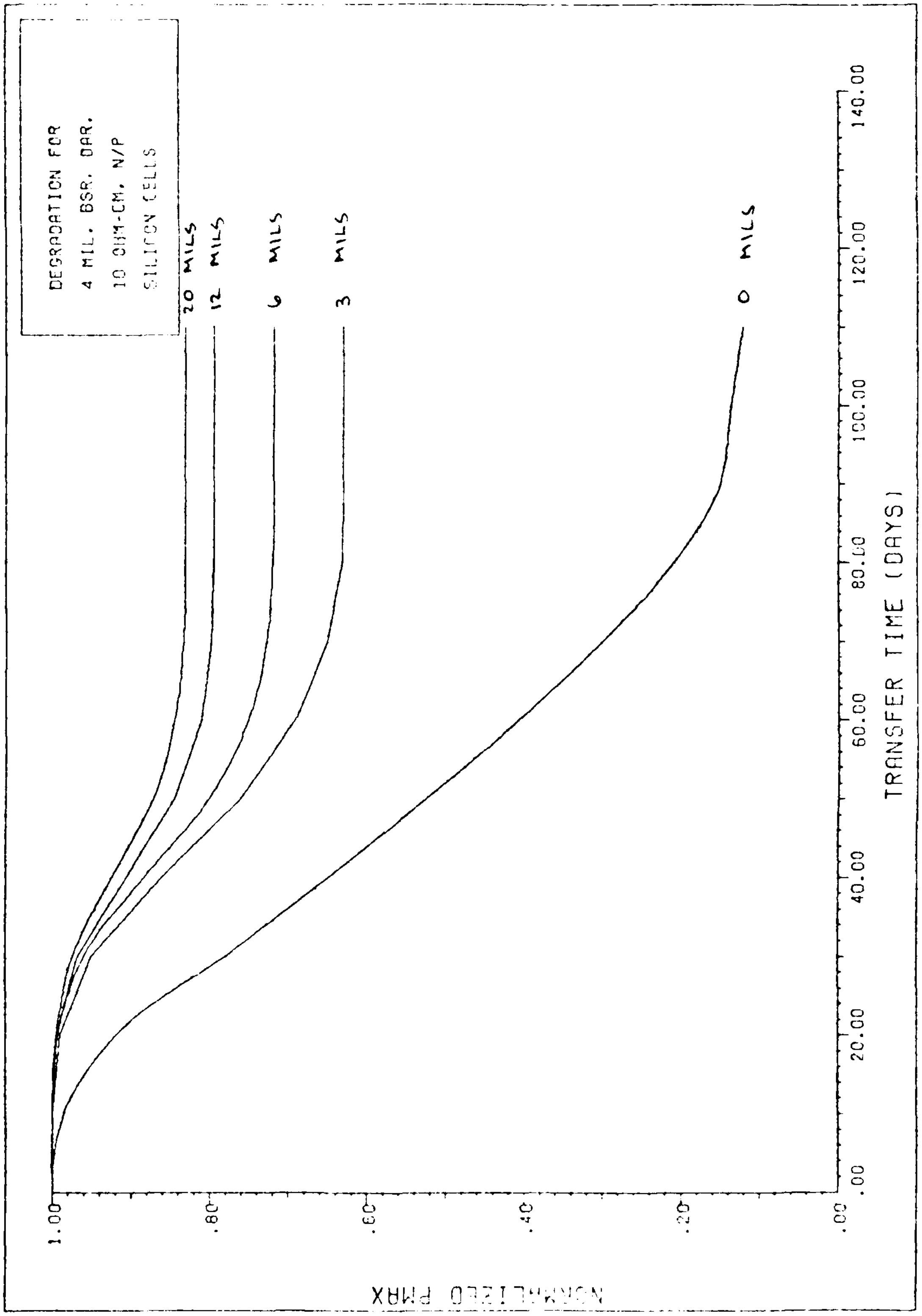


PROTON FLUENCE FOR 110 DAY TRANSFER AT 55 DEGREES

Figure 6-2



SOLAR ARRAY DEGRADATION: 15_ DAYS, 28.5° TO 55°



SOLAR ARRAY DEGRADATION: 110 DAYS, NO DELTA I, 55°

Figure 6-4

recalculated. These values are found in Figures 6-5 - 6-8.

Upon comparing the 70 day fluence levels with the 150 and 110 day fluence levels, it was noted that there was not a significant difference between them. Table 6-1 clearly shows that they all were of the same order of magnitude.

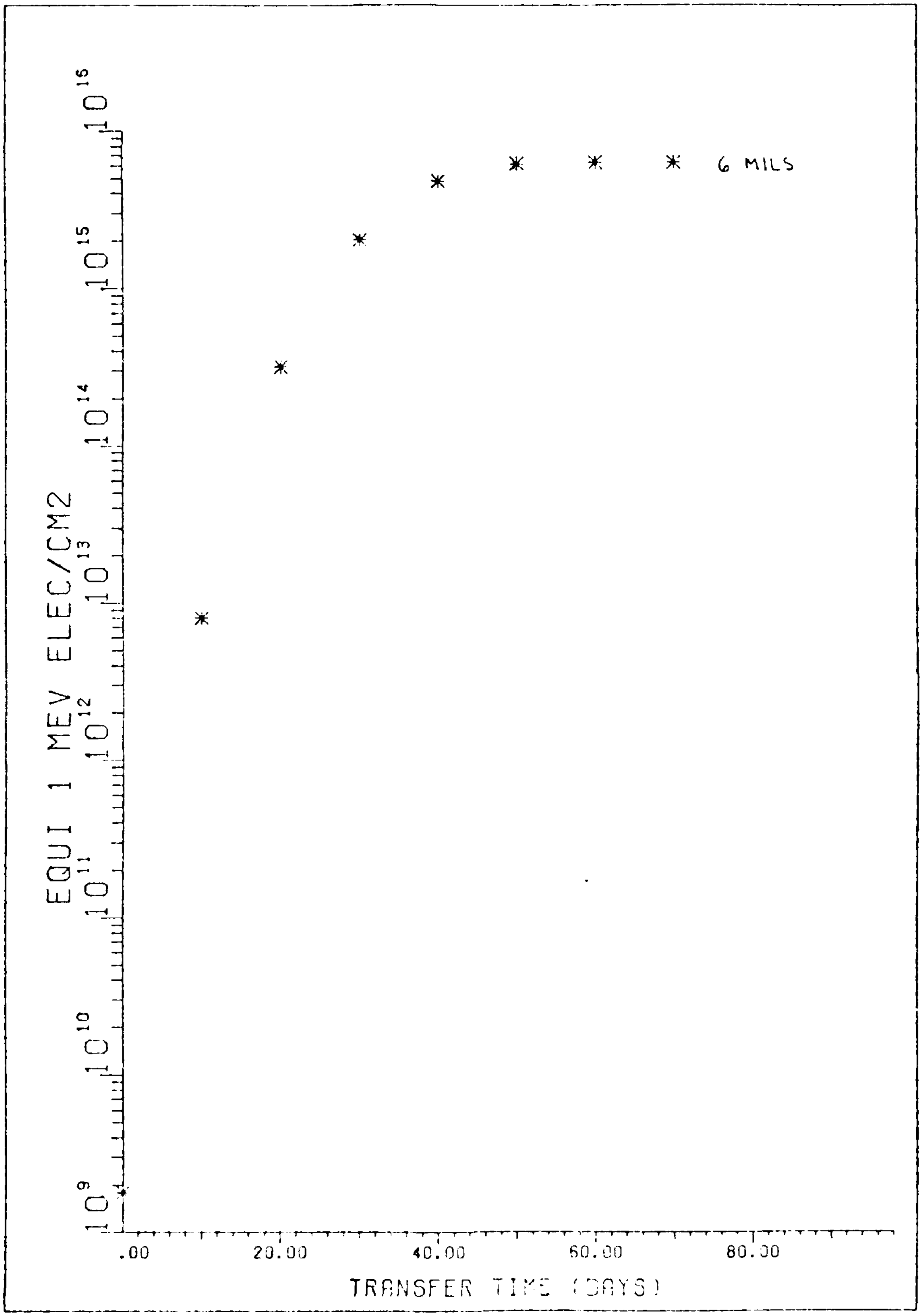
Table 6-1. Fluence Levels for 6 mil Covers

Transfer Orbit		Cumulative Fluence (1 MeV/cm2)
150 days: 28.5' to 55'	-	9.05898 E+15
110 days: 55'	-	4.52520 E+15
70 days: 28.5' to 55'	-	6.31422 E+15
70 days: 55'	-	2.33426 E+15

The difference in triptimes did not significantly affect the fluence level. Therefore, when it became necessary to determine the fluence level for the return trip, the same value as for the outbound trip was used. Thus it was possible to estimate cumulative effects from several trips thru the Van Allen belts. A sample of this is found in Figure 6-9.

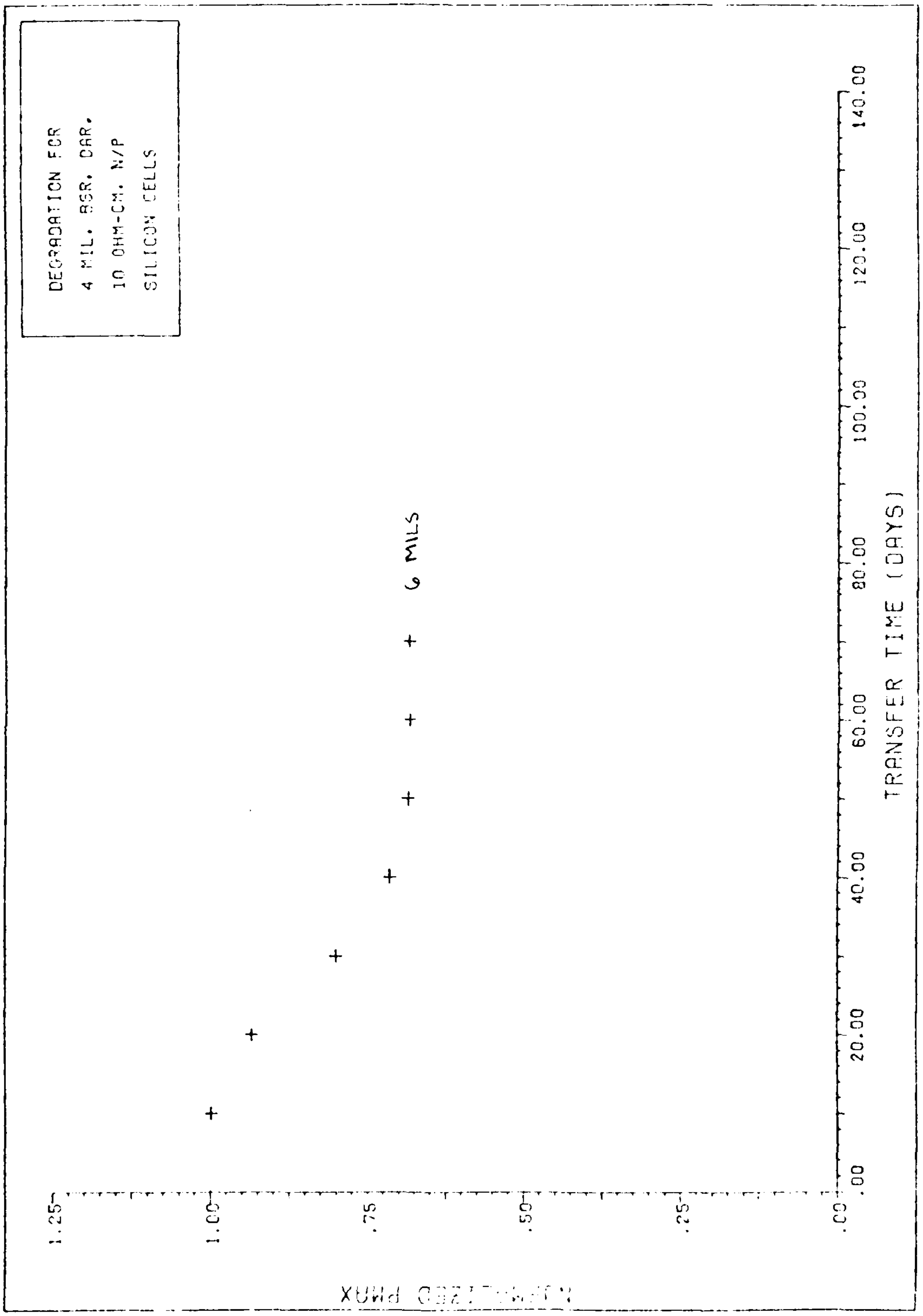
INPUT PARAMETER VALUES

The values used for the input parameters are presented below. References are listed for the input parameters for which a method of derivation is not presented in either the methodology section or the Appendices. Parameters for which a range of values exist are followed by a value in parenthesis which represents the baseline value.



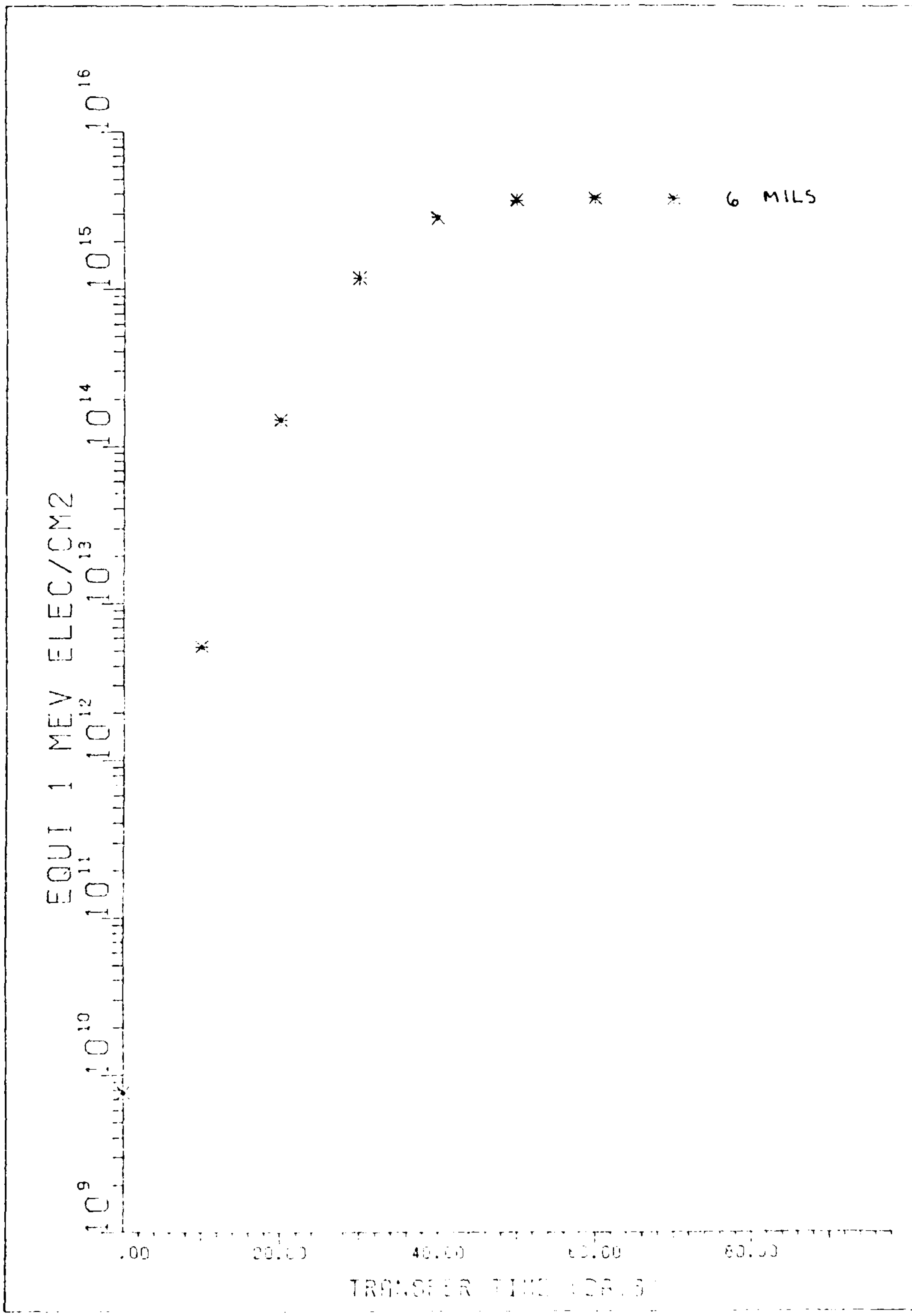
PROTON FLUENCE FOR 70 DAY TRANSFER: 28.5° TO 55°

Figure 6-5



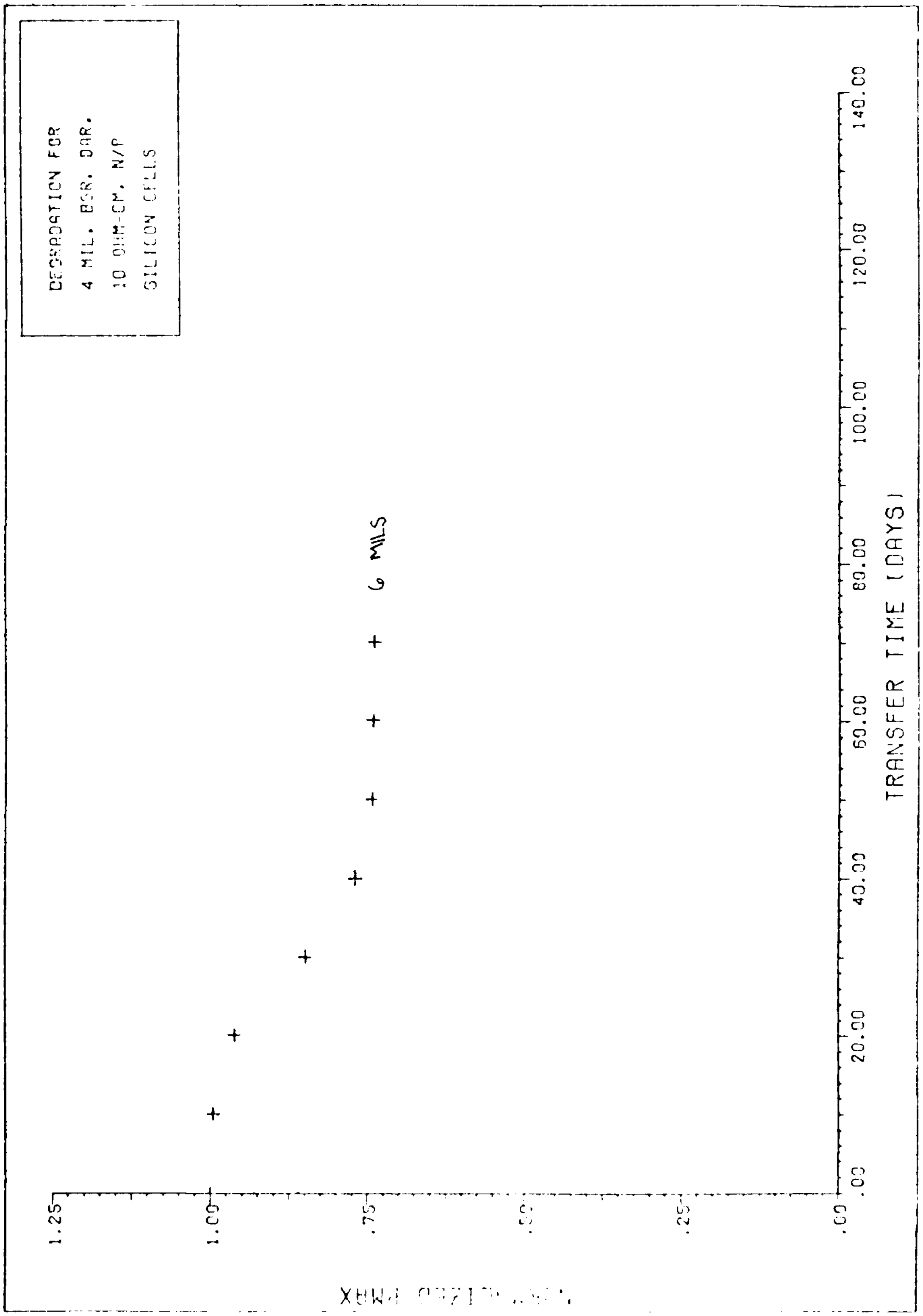
SOLAR ARRAY DEGRADATION: 70 DAYS, 28.5° TO 55°

Figure



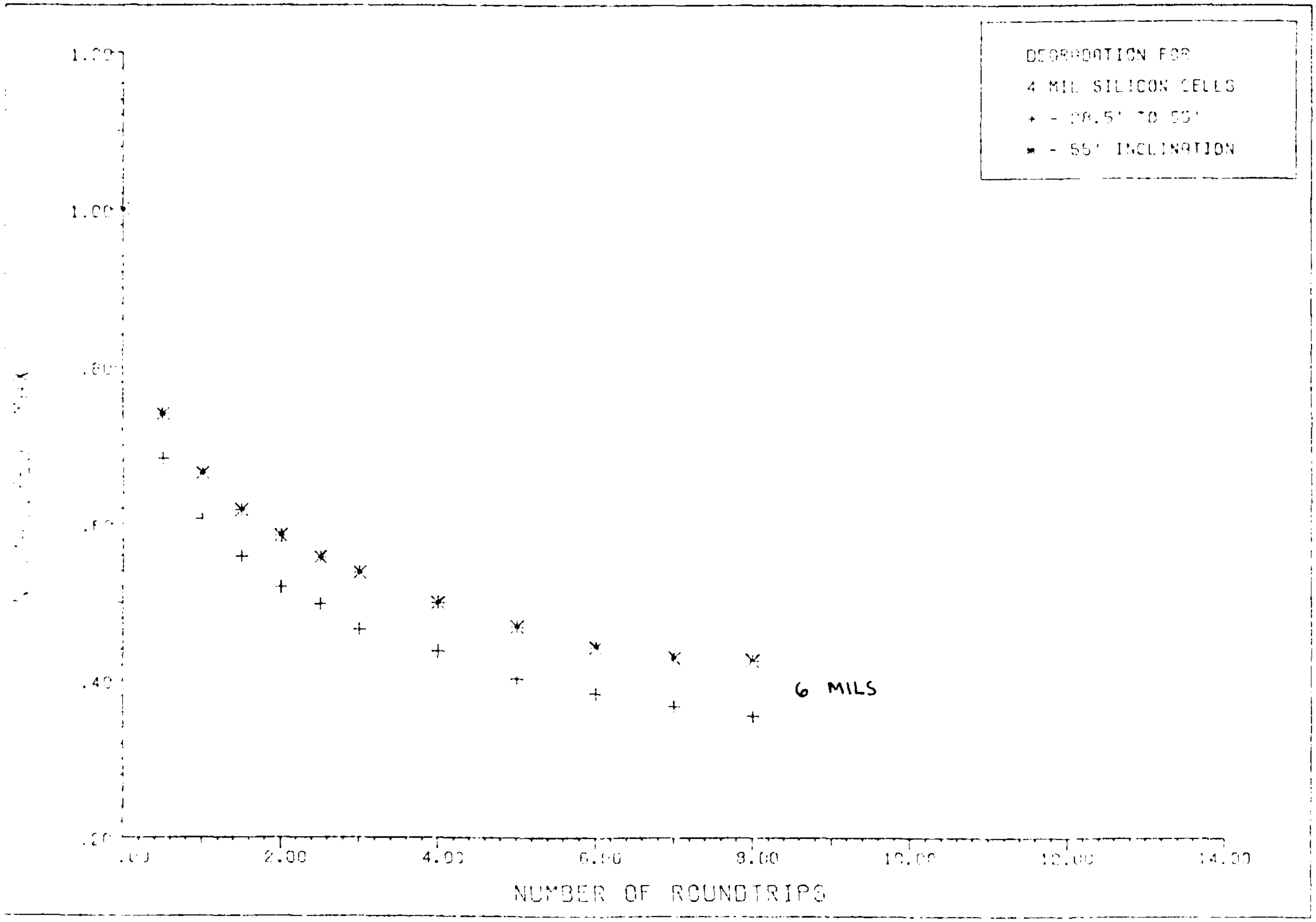
PROTON FLUENCE: [illegible] DAY TRANSFER AT 55° INCLINATION

Figure 6-7



SOLAR ARRAY DEGRADATION: 70 DAYS, 55° INCLINATION

Figure [illegible]


DEGRADATION FOR
4 MIL SILICON CELLS
+ - 28.5° TO 55°
* - 55° INCLINATION
6 MILS
NUMBER OF ROUNDTRIPS
.00
2.00
4.00
6.00
8.00
10.00
12.00
14.00
.20
.40
.60
.80
1.00
1.20

SOLAR ARRAY [illegible]

[illegible]

SOLAR ARRAY.

Before final input values could be determined, it was necessary to decide on a solar array shield thickness to use. This was done by comparing the triptimes for EOTVs that differed only in array shield thickness. It turned out that in all cases examined, the EOTV with a 6 mil array shield had the lowest triptime. Arrays with thicker shields degraded less and were therefore smaller; however, the extra weight of this additional shielding negated the gains of the lesser degradation. Inversely, arrays with less shielding degraded much more and the weight due to the increased size was greater than the savings from the thinner shielding.

The following tables summarize the values used for the arrays with 6 mils of microsheet covers.

Table 6-2. Degradation Values (R)

	28.5' to 55'	55'
Silicon Flatplate Array		
a) 1 roundtrip	.430	.357
b) 7 roundtrips	.632	.570
c) 1 outbound	.317	.260
Gallium Arsenide Concentrator	.05	.05

Table 6-3. Specific Mass and Specific Cost

	Present	1990's
Silicon Flatplate Arrays:		
ASA (kg/KW)	15.15	7.508
GSA ($/W)	150	150
Gallium Arsenide Concentrators:		
ASA (kg/KW)	N/A	18.36
GSA ($/W)	N/A	150-300 (225)

Values for the specific mass are derived in Appendix The values for the specific costs and the degradation va for the Gallium Arsenide (Ga-As) concentrators are estim made by Dr. Pat Rahilly of the Air Force Wright Aeronauti Laboratories at Wright Patterson AFB.

ELECTRIC PROPULSION SYSTEM.

The values in the following table were compiled from following sources listed in the Bibliography: 16, 36, 40, 66. All of these values describe a representative system w a 30 cm ion engine with a constant thrust level of 129 The system includes the necessary power processing units, support structure, the propellant tank and lines, and radiator.

Table 6-4. Engine System Parameters

	Present	1990'
1. Mercury (Hg)		
ISP (seconds)	2900	3000
Input Power (KW)	3.06	2.283
Mass (kg)	51	40
Cost ($/system)	688500	35000
Efficiency	.603	.83
2. Argon (Ar)		
ISP (seconds)	6270	6000
Input Power (KW)	5.785	4.515
Mass (kg)	55	45
Cost ($/system)	742500	39375
Efficiency	.68	.84
3. Xenon (Xe)		
ISP (seconds)	4560	3500
Input Power (KW)	3.98	2.67
Mass (kg)	51	40
Cost ($/system)	688500	35000
Efficiency	.603	.83

The number of engines used was varied from 1 to 8 in single unit increments and then up to 40 in 4 unit increments. The guidance and control package was estimated to weigh 50 kg and cost $ 1 MIL.

PAYLOAD.

The exact weight of the Block 3 satellite is not known. Captain Sponable of SD/YEZ, estimated that it would be around 3000 pounds. Therefore a baseline payload weight of 1500 kg was used and in the sensitivity studies, this was varied from 1000 kg. to 2000 kg.

TRAJECTORY.

The only variable that was not constant was the initial inclination. This was either 28.5 degrees for the nominal shuttle launch orbit or 55 degrees for a shuttle launch that would not require the EOTV to perform an inclination change.

The values used for the other variables were:

a) initial altitude - 200 km

b) final altitude - 20,200 km

c) final inclination - 55 degrees

d) drag penalty factor - .001

e) time penalty factor for engine restart - .18

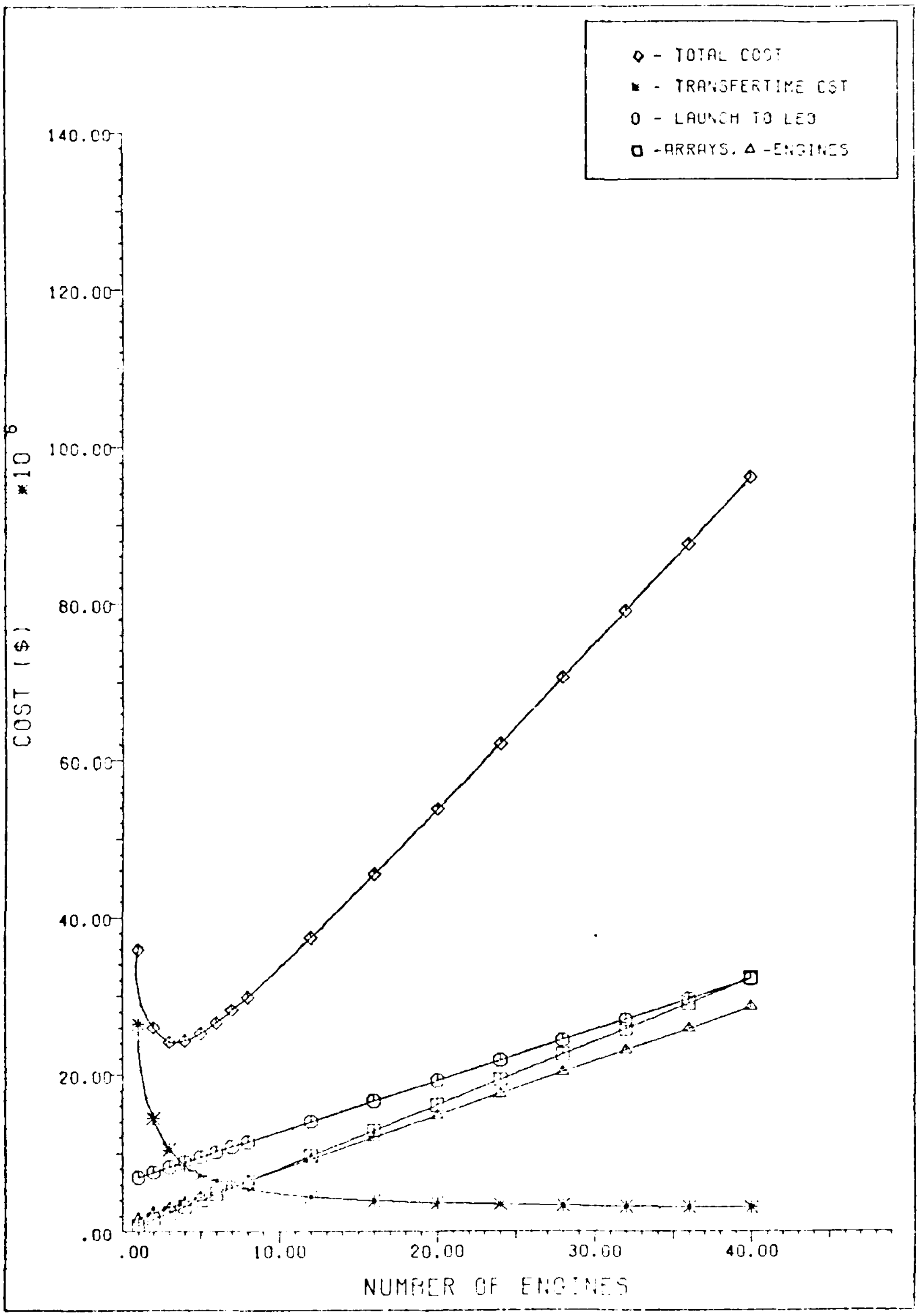
f) penalty factor for time in shadow - .025

The value for the drag penalty factor was obtained from estimates used in the Boeing system model (26). The derivations of the values for e) and f) above are included in the Appendix F.

RESULTS

The first runs were made to compare the present technology system using three different fuels: mercury, xenon, and argon. The initial orbit inclination was 28.5 degrees. Before looking at the results of these runs, it is necessary to first examine the output of the model. For this analysis, the output for the system using mercury as the propellant will be examined.

Consider the plot of Number of Engines versus Cost (Figure 6-10). The cost curves for the solar arrays, engines, and launch to LEO are all linear and all increase as the number of engines increases. The non-negligible transfer time costs (operations costs) decrease as the number of engines increases because more engines mean a shorter transfer time. This decrease however is nonlinear, with the greatest change taking place at low engine numbers. With fewer engines, triptimes are very long and the transfer time costs are very significant compared to the hardware and launch costs. The sum of all these costs produces a total cost curve that decreases initially, bottoms out, and then increases. At low engine numbers, the increase in hardware and launch costs associated with adding an engine are less than the savings produced by the reduced triptime and the resulting decrease in transfer time costs. However, as the triptimes continue to decrease, the transfer time costs become less significant and the savings gained by adding more engines is much less than the increase in hardware and launch



ENGINE NUMBER VS TOTAL COST: MERCURY ION ENGINES

Figure 6-10

costs.

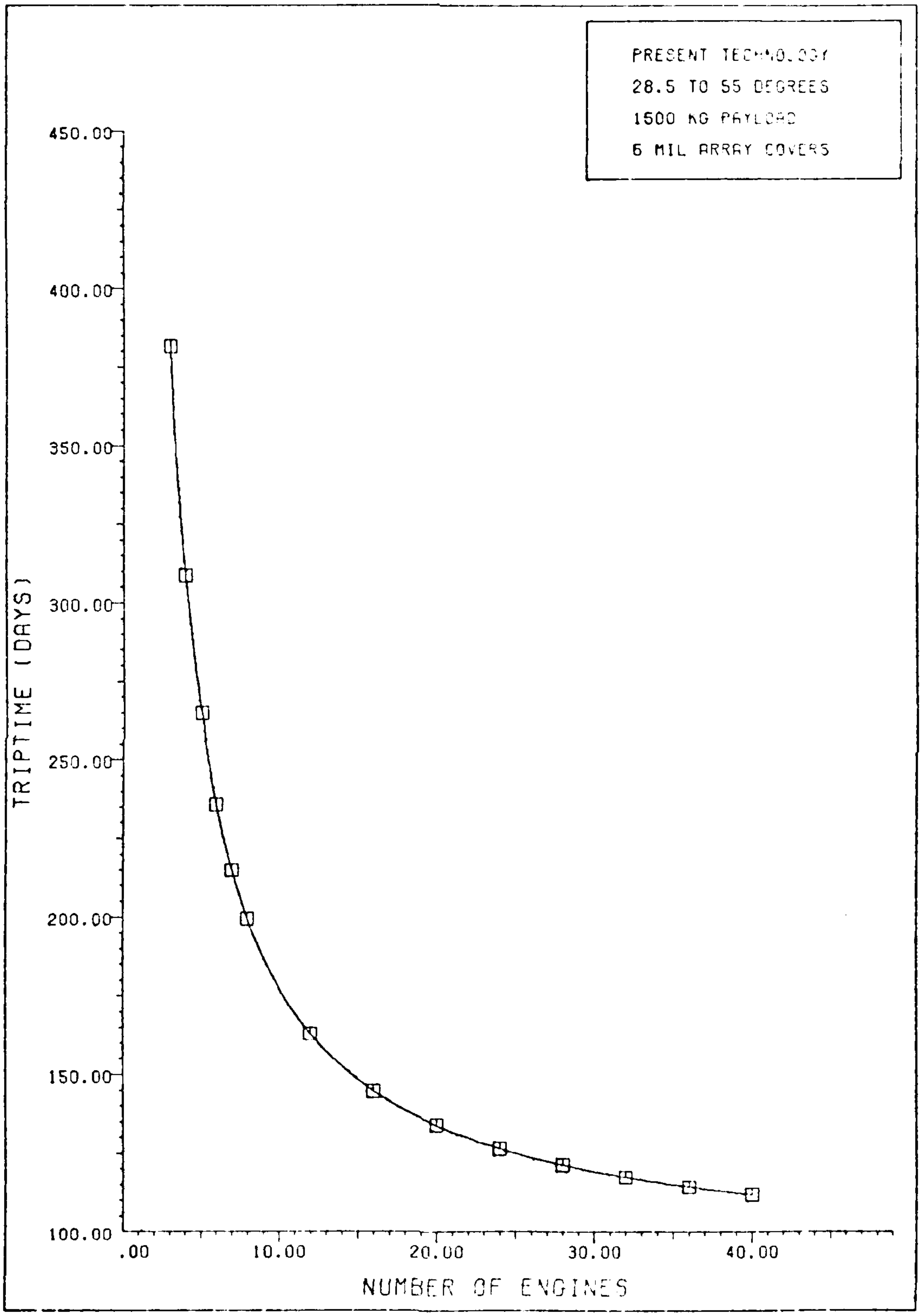
The plot of the Number of Engines versus Triptime (Figure 6-11) shows the significant influence low engine numbers have on triptime. The curve begins to level off at higher engine numbers because the thrust gained by adding an engine becomes less significant than the increase in the weight of the added engine and the extra solar array. The higher total weight causes the effective increase in acceleration to be smaller; therefore, the decrease in triptime also becomes smaller.

The Triptime versus Cost plot (Figure 6-12) further illustrates the tradeoff between the two factors of interest. The triptime corresponding to the minimum cost (the lowest point on the curve) can easily be read. If this triptime is greater than 90 days, the change in cost to get it down to, or below 90 days is easily found.

Returning now to the results of the first runs, Figure 6-13 and Table 6-5, one concludes that mercury produces the least cost for a given triptime. However, the minimum cost point has a triptime of 381 days which is well past the acceptable limits.

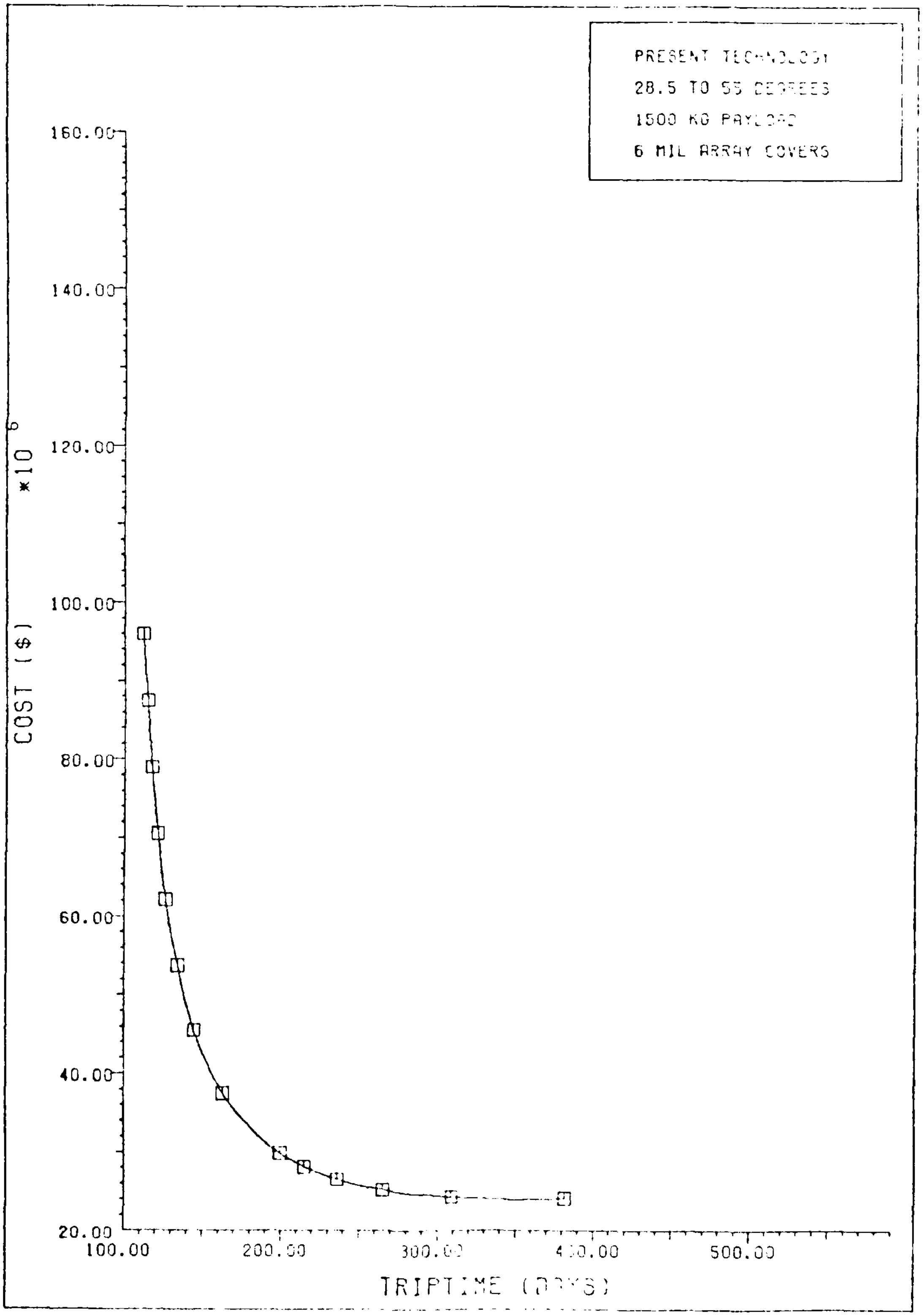
Table 6-5. Present Technology Minimum Cost Points

	Hg	Xe	Ar
# of Engines	3	3	3
PNOM (KW)	16.105	20.947	30.447
Outbound Time (days)	381.52	380.86	402.10
Cost ($ Millions)	24.15	24.46	26.92



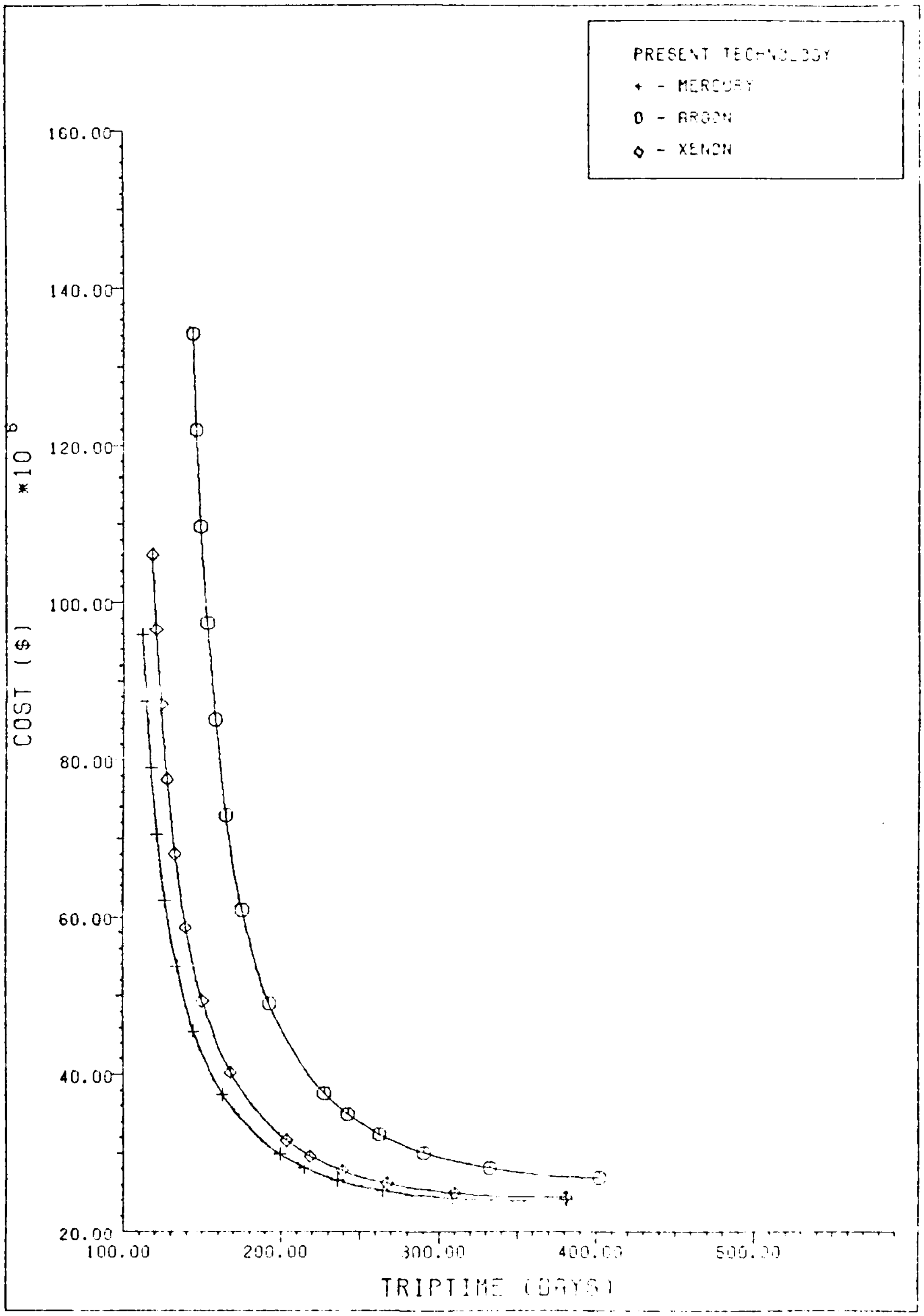
ENGINE NUMBER VS TRIPTIME FOR MERCURY ION ENGINES

Figure 6-11



TRIPTIME VS TOTAL COST FOR MERCURY ION ENGINES

Figure [illegible]



TRIPTIME VS COST FOR VARIOUS ENGINE PROPELLANTS

Figure 6-13

The next set of runs compared the 1990's technology systems again using the three different fuels and starting at 28.5 degrees. The results, shown in Figure 6-14 and Table 6-6, show improvement in all three systems with mercury remaining the system with the least cost for a given triptime.

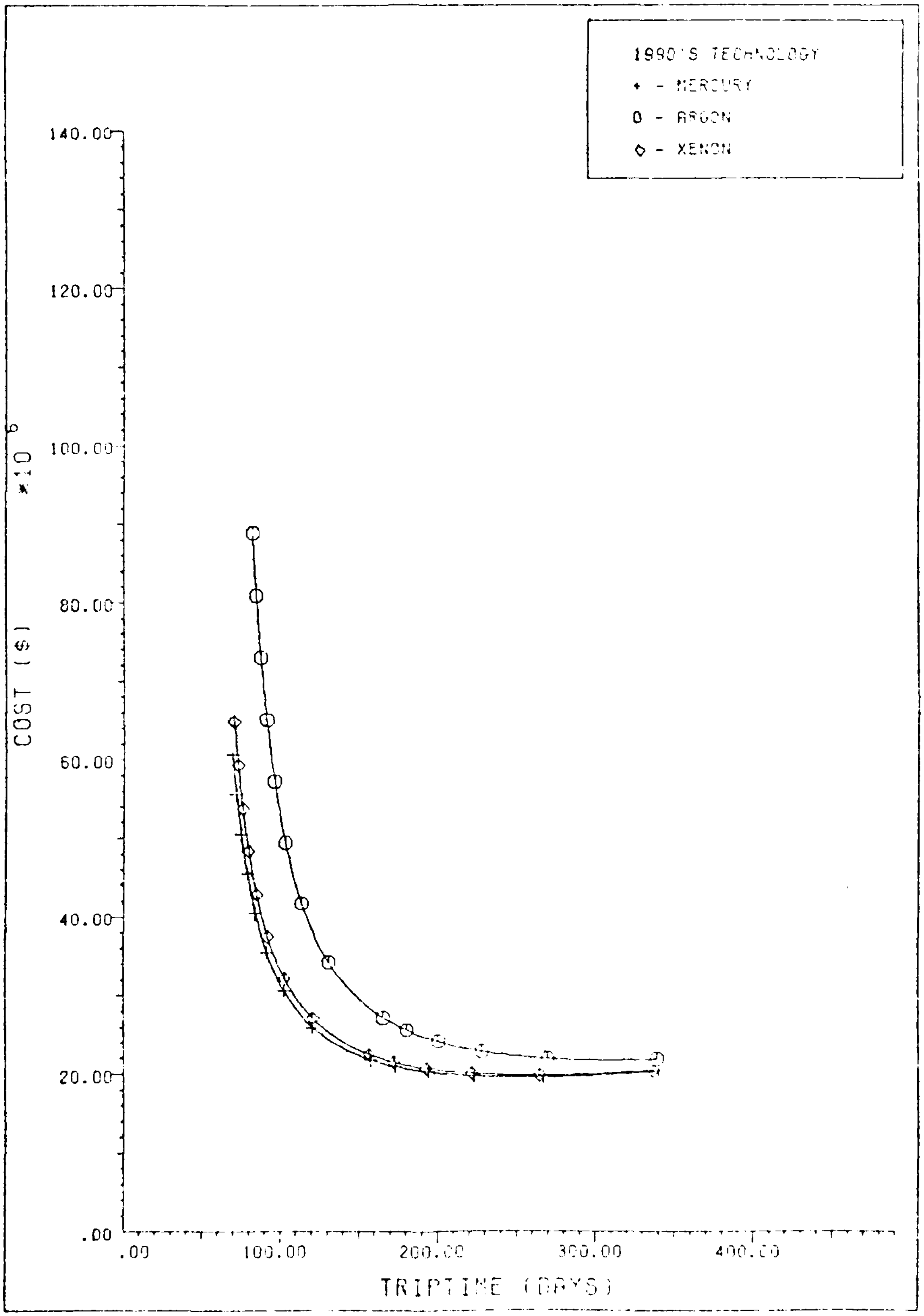
Table 6-6. 1990's Technology Minimum Cost Points

	Hg	Xe	Ar
# of Engines	4	4	3
PNOM (KW)	16.021	18.737	23.763
Outbound Time (days)	266.98	264.52	339.101
Cost ($ Millions)	19.70	19.89	21.94

Table 6-7. 1990's Technology System Data : 20 - 32 Engines

	Hg	Xe	Ar
20 Engines			
PNOM	80.105	93.684	158.421
Outbound Time (days)	91.45	92.01	103.121
Cost ($ Millions)	35.51	37.50	49.46
24 Engines			
PNOM	96.126	112.421	190.105
Outbound Time (days)	84.14	84.82	96.18
Cost ($ Millions)	40.46	42.89	57.26
28 Engines			
PNOM	112.147	131.158	221.790
Outbound Time (days)	78.92	79.68	91.22
Cost ($ Millions)	45.48	48.33	65.12
32 Engines			
PNOM	128.168	149.895	253.474
Outbound Time (days)	75.00	75.83	87.50
Cost ($ Millions)	50.52	53.81	73.02

Table 6-6 shows that the minimum cost points still have very long triptimes. It appears that in order to reduce the



TRIPTIME VS COST FOR VARIOUS ENGINE PROPELLANTS

Figure 6-14

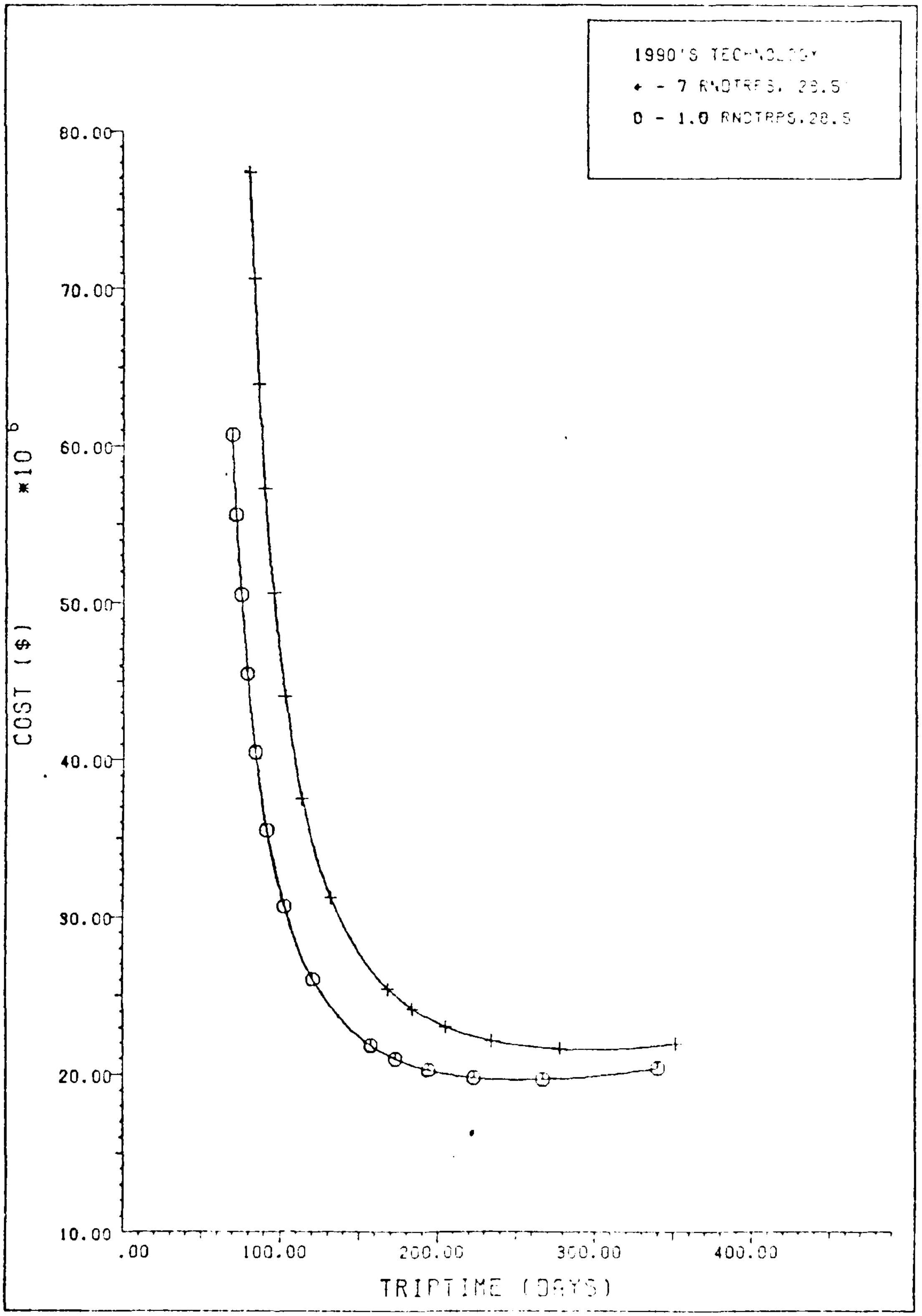
triptimes to the acceptable level, an operating point other than the minimum cost point will have to be used. From Table 6-7 it appears that it will be necessary to use from 20 to 32 engines in order to reduce the triptime below 90 days.

Because mercury was the propellant with the best results, all further analysis was done using only this propellant.

Preliminary runs of the model using input parameter values found in the available literature had indicated that the best triptimes achievable were approximately 150 days for an EOTV starting at 28.5 degrees and 110 days for an EOTV starting at 55 degrees. These were the basis for the calculations for the fluence levels and the degradation factors for the solar arrays. With the new data indicating achievable triptimes of 70 - 90 days, it became necessary to recalculate the fluence levels and degradation factors. The results of these calculations were presented earlier in Figures 6-5 - 6-8.

The engines which have a predicted lifetime of 15000 to 20000 hours could now be expected to last for approximately seven roundtrips. New degradation factors were obtained from Figure 6-9 and new runs were made using the 1990's technology system parameters to see what changes this produced.

Figure 6-15 and Table 6-8 indicate that the increased degradation produced significant changes. Not only did triptimes increase by 11 - 12 days, but costs also went up



EFFECT OF INCREASED DEGRADATION ON TRIPTIME & COST

Figure 6-5

$10 - 11 Mil. The most significant problem however, was the increase in PNOM which corresponds to an increase in solar array size.

Table 6-8. Effects of Increased Degradation

Degradation Factor:	7 Roundtrips	1 Roundtrip
24 Engines		
PNOM	148.891	96.126
Outbound Time (days)	95.344	84.14
Cost ($ Millions)	50.60	40.46
28 Engines		
PNOM	173.707	112.147
Outbound Time (days)	90.12	78.92
Cost ($ Millions)	57.25	45.48
32 Engines		
PNOM	198.522	128.168
Outbound Time (days)	86.20	74.996
Cost ($ Millions)	63.93	50.52

Using a 4m by 32m array as standard (this is the size of the array deployed on the initial flight of the shuttle Discovery), it was possible to relate PNOM to number of arrays.

The specific power for the 1990's technology Space Frame Array (see Appendix E) is .017298 KW/sq.ft. The 4m by 32m array contains 1355.9 sq ft and therefore is capable of producing 23.45 KW. Table 6-9 shows the number of arrays needed by the EOTV.

Looking at these systems in terms of number of solar arrays required made it very clear that none of them, even those using only 1 roundtrip degradation, was feasible. They simply required too many arrays.

Table 6-9. Solar Array Requirements : 28.5 Degrees

Degradation Factor:	7 Roundtrips	1 Roundtrip
24 Engines		
PNOM (KW)	148.891	96.126
# of Arrays	6.4	4.1
28 Engines		
PNOM (KW)	173.701	112.147
# of Arrays	7.5	4.8
32 Engines		
PNOM (KW)	198.522	128.168
# of Arrays	8.5	5.5

Since the degradation at 55 degrees inclination is less, a run was made to see if the array requirement was reduced enough to be feasible. Table 6-10 shows that although there was some improvement, except for the 12 engine system, all others still required too many arrays. For this reason, it was decided that flat plate Silicon cell arrays are not suitable for a reusable EOTV.

Table 6-10. Solar Array Requirements : 55 Degrees

# of Engines	PNOM (KW)	# of Arrays
12	63.712	2.7
16	84.949	3.6
20	106.186	4.5
24	127.423	5.4
28	148.660	6.3

Having ruled out the use of flat plate silicon cell arrays for a reusable system, the only other alternative considered was the use of Gallium Arsenide concentrators. Runs were made to evaluate their performance at initial inclinations of 28.5 and 55 degrees. Figure 6-16 and Table

AD-A152 021 ANALYSIS OF ORBIT TRANSFER VEHICLES FOR GPS BLOCK 3 SATELLITES(U) AIR FORCE INST OF TECH WRIGHT-PATTERSON AFB OH SCHOOL OF ENGINEERING D P BOYARSKI ET AL. 2/2

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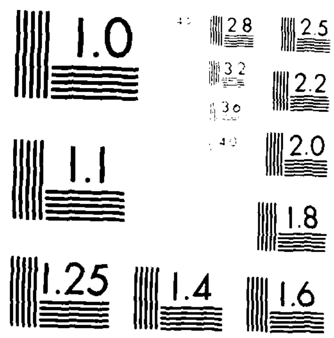


MICROCOPY RESOLUTION TEST CHART

6-11 contain the results.

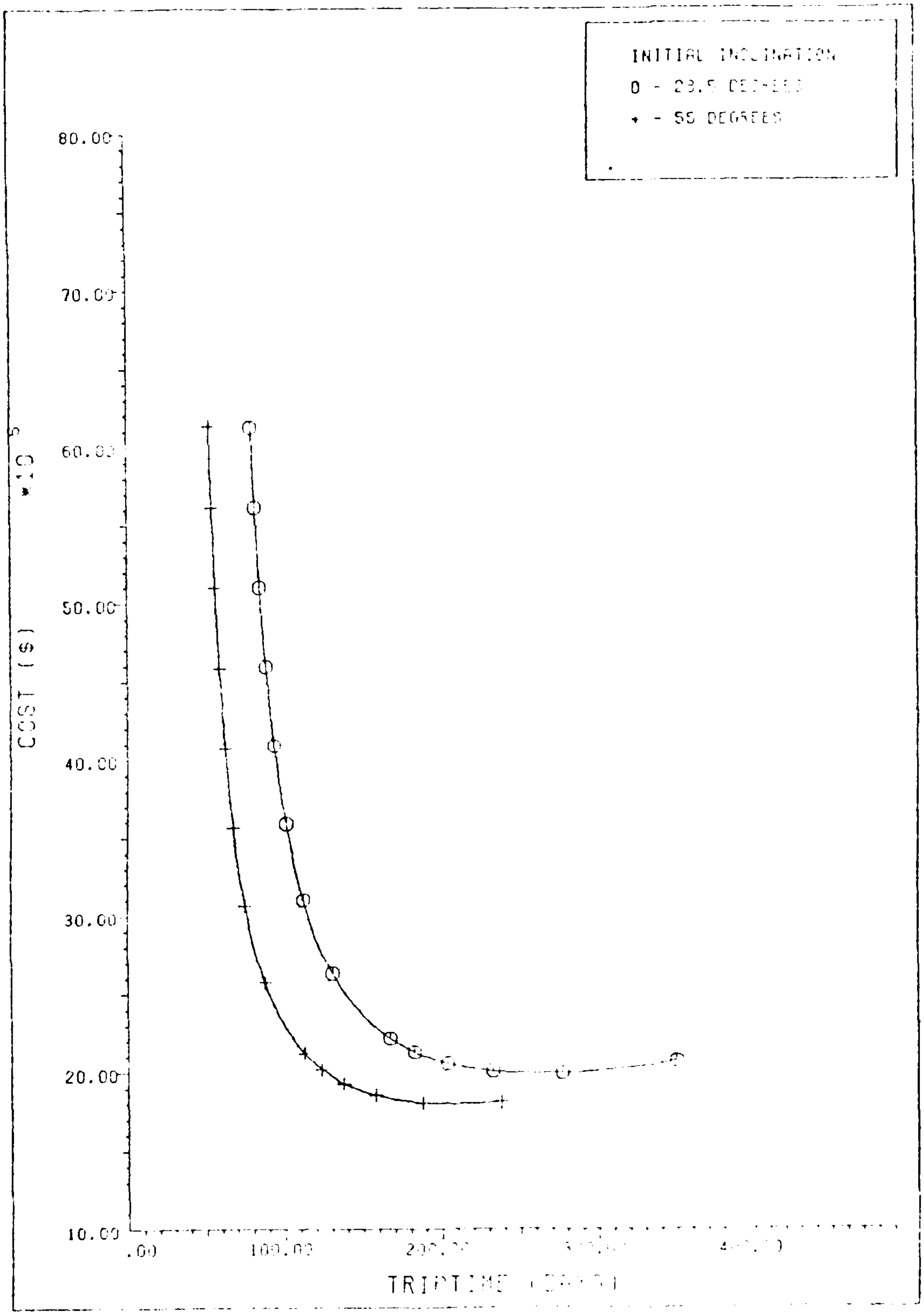
Table 6-11. 1990's Ga-As Concentrators

Initial Inclination:	28.5'	55'
12 Engines		
PNOM	28.84	28.84
Outbound Time (days)	130.25	87.45
Cost ($ Millions)	26.36	25.77
# of Arrays	1.01	1.01
16 Engines		
PNOM	38.45	38.45
Outbound Time (days)	111.96	74.89
Cost ($ Millions)	31.04	30.64
# of Arrays	1.35	1.35
20 Engines		
PNOM	48.06	48.06
Outbound Time (days)	100.99	67.35
Cost ($ Millions)	35.92	35.66
# of Arrays	1.69	1.69
24 Engines		
PNOM	57.68	57.68
Outbound Time (days)	93.68	62.33
Cost ($ Millions)	40.91	40.74
# of Arrays	2.02	2.02
28 Engines		
PNOM	67.29	67.29
Outbound Time (days)	88.45	58.74
Cost ($ Millions)	45.95	45.86
# of Arrays	2.36	2.36

Using the same array size previously mentioned and with the specific power for the gallium arsenide concentrators at .021 KW/sq ft, the array sizes became very reasonable.

It was noted that by going to a 55 degree initial inclination, the triptime was reduced by approximately 33% while costs remained virtually equal, in spite of the increased launch costs.

Comparing this to the 7 roundtrip Silicon array system



1990'S MERCURY ENGINES [illegible]

Figure [illegible]

at 55 degrees initial inclination, the Ga-As system triptime was only 2 days greater, cost was $3-7 Mil less, and the array size was about one third. The conclusion is that Ga-As concentrators are the best choice of power source for a reusable system.

A non-reusable system was another option that needed to be examined. Since the silicon arrays were the lighter of the two options, they were used. The degradation factors were adjusted to reflect only one passage thru the Van Allen belts.

The results of the runs (Figure 6-17 and Table 6-12) indicated that a non-reusable system starting at 28.5 degrees would require large arrays. However, at 55 degrees inclination, a system with 12 engines would have a triptime of 78.18 days at a cost of $23.34 Mil. As long as triptimes up to 90 days remain acceptable, this system could be a viable candidate.

The results thus far have indicated that :

1) Due to the triptime and solar array size constraints, it is necessary for the EOTV to have an initial inclination of 55 degrees.

2) This initial inclination requirement does not cost more. The increase in launch cost is balanced by the reduction in transfer time costs.

3) For a reusable system, Gallium Arsenide concentrators are the power source of choice.

4) For a non-reusable system, Silicon arrays are

the power source of choice.

5) Mercury is the propellant that produces the least cost system; however, if environmental considerations make its use unacceptable, then Xenon should be used. The changes in cost and time would be very small.

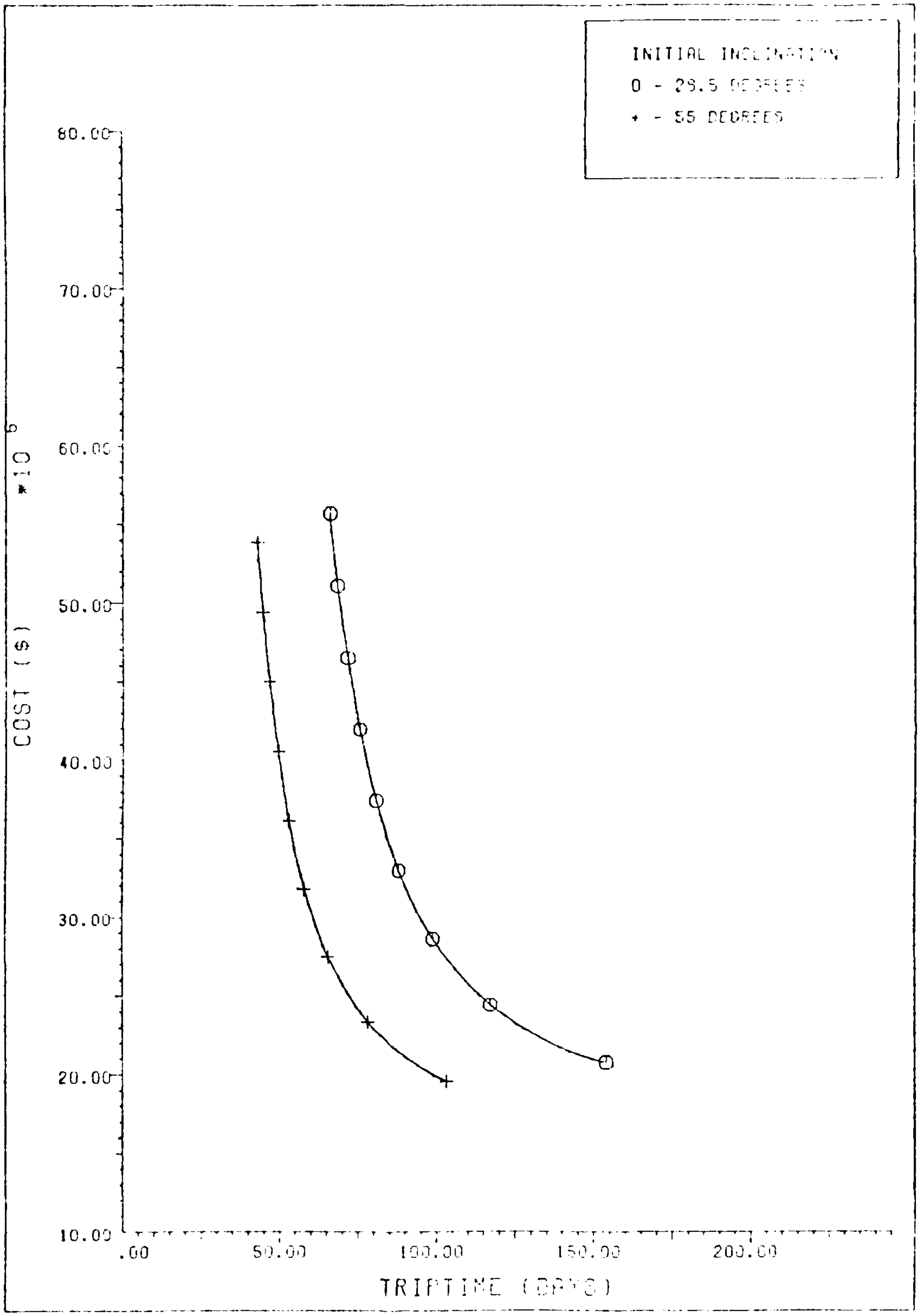
Table 6-12. Non-reusable Silicon Array EOTVs

Initial Inclination:	28.5'	55'
12 Engines		
PNOM	40.11	37.02
Outbound Time (days)	117.33	78.18
Cost ($ Millions)	24.43	23.34
# of Arrays	1.7	1.6
16 Engines		
PNOM	53.48	49.36
Outbound Time (days)	99.05	65.62
Cost ($ Millions)	28.59	27.49
# of Arrays	2.3	2.1
20 Engines		
PNOM	66.85	61.70
Outbound Time (days)	88.07	58.08
Cost ($ Millions)	32.95	31.78
# of Arrays	2.85	2.6
24 Engines		
PNOM	80.22	74.04
Outbound Time (days)	80.76	53.06
Cost ($ Millions)	37.41	36.14
# of Arrays	3.42	3.16

SYSTEMS EVALUATION

Having determined which power systems are best suited for use as reusable and non-reusable EOTVs, their life cycle costs were analyzed to obtain a per satellite deployment cost.

For the non-reusable system, no further manipulation of the output was necessary. The costs listed in Table 6-12 are



NONREUSABLE 1400'S MERCURY ENGINES WITH SI ARRAYS

Figure 6-27

the total costs to place a satellite into its proper orbit. The best alternative here is the 12 engine system with mercury propellant and an initial inclination of 55 degrees. The per satellite cost using this system is $ 23.34 million.

For the reusable systems, the costs for launches other than the first still need to be determined. What follows is an example of how these costs were calculated. The system being evaluated is the 1990's technology system using mercury propellant, Ga-As concentrator arrays and 12 engines. Figure 6-18 is the computer output for this system.

```
FOR THE OUTBOUND TRIP:
ISP =  3000   NUMBER OF ENGINES = 12
POWER REQUIRED = 27.396
PNOM = 28.8379
CSA =  6.48853E+06            MSA = 529.464
CEPS =  5.2E+06               MEPS = 530
DELI = 0
CP =  5791.85                 MP = 386.123
N = .83
TRANSFER TIME IS 2098.69 HOURS OR 87.4455 DAYS
THRUST TIME = 2047.5
CTT =  2.39577E+06
CETO =  1.16771E+07
CM =  2.57672E+07             MT = 3483.58
COST FACTOR =  17178.1

FOR THE RETURN TRIP:
MPR = 150.931                                    CPR = 2263.97
RETURN OPS COST = 936479
TOTAL COST FOR RETURN = 938743
RETURN TIME IS 820.356 HOURS OR 34.1815 DAYS
THRUST TIME = 800.347

ROUNDTRIP TIME IS 2919.05 HOURS OR 121.627 DAYS
TOTAL THRUST TIME =  2847.85
```

Figure 6-18. Computer Output for 12 Engine System

Using this data and the launch cost equations presented earlier, the following results were obtained.

First Roundtrip :		$ 26,705,943
Subsequent Roundtrips:		
a) CETO : payload =		1500.000 kg
	roundtrip fuel =	537.054 kg
	TOTAL	2037.054 kg
	CETO :	$ 6,828,260
b) outbound CTT :		$ 2,395,770
c) outbound CP :		$ 5,792
d) return costs :		$ 936,479
	TOTAL	$ 10,166,301
Last Outbound Trip :		
CETO + CTT + CP =		$ 9,229,822

The first roundtrip cost is much higher than the rest because it includes the nonrecurrent cost of bringing the EOTV into orbit as well as the cost of the EOTV itself. On subsequent roundtrips, the shuttle only has to bring the new payload and more fuel up to LEO. The reason for having a separate cost for the last outbound trip is that as the EOTV reaches the end of its useful life (engine burn out), the EOTV will not be brought back to LEO after deploying its last satellite. Therefore, this last deployment will not incur the return costs.

Having calculated these costs, it was then necessary to calculate the useful life of the EOTVs. As alluded to

earlier, engine burnout determines the EOTV's useful life. Since the estimates for engine life were from 15000 to 20000 hours, it was decided to run the calculations at each end of the span.

From Figure 6-18, the total thrust time for one roundtrip is 2847.85 hours. This means that a 15000 hour engine would be good for 5.3 roundtrips (RTs) and a 20000 hour engine for 7.0 roundtrips. In terms of actual usage, 5.3 roundtrips would be 4 roundtrips plus 1 last outbound trip. It is not possible to make 5 roundtrips plus 1 last outbound trip because the outbound thrust time of 2047.5 hours or .72 roundtrip exceeds the remaining time available on the engines. Therefore after the fifth outbound trip, the EOTV is considered burned out.

Because the roundtrip time is 121.6 days, a single EOTV would not be able to handle the yearly requirement of four deployments per year (one every 90 days). Two EOTVs would be necessary. These EOTVs would be able to deploy a total of 10 satellites or 2.5 years worth. The costs to do this would be as follows:

2 initial RTs	:	$ 53,411,886
6 middle RTs	:	$ 60,997,806
2 last outbounds	:	$ 18,459,644
		$ 132,869,336

or $ 13,286,934 per satellite

Similarly, for an engine life of 20000 hours, two EOTVs would deploy 14 satellites or 3.5 years worth. The costs

would be:

2 initial RTs	: $ 53,411,886
10 middle RTs	: $ 101,663,010
2 last outbounds	: $ 18,459,644
	$ 173,534,540

or $ 12,395,325 per satellite

To see if the decrease in triptime achieved by adding engines affected the costs, these calculations were repeated for 16, 20 and 24 engines. The results are found in Tables 6-13 and 6-14.

Table 6-13. Per Satellite Deployment Costs ($ Million)

Engines	Triptime (Days)		Engine Life	
	Outbound	Roundtrip	15000 hrs	20000 hrs
12	87.4	121.6	13,286,934	12,395,325
16	74.9	108.7	13,573,161	12,719,674
20	67.4	100.9	14,519,329	13,462,481
24	62.3	95.7	14,816,076	13,855,986

Table 6-14. Useful Life / Satellites Deployed (2 EOTVs)

Engines	15000 hrs	20000 hrs
12	2.5 yrs / 10	3.5 yrs / 14
16	3.0 yrs / 12	4.0 yrs / 16
20	3.0 yrs / 12	4.0 yrs / 16
24	3.5 yrs / 14	4.5 yrs / 18

These results indicate that although more engines mean

shorter triptimes and more trips, the additional cost of the extra engines and solar arrays exceed the benefits the extra trips produce in cost averaging. Thus the 12 engine system is the best (cost-wise) to use.

For the comparison with the nuclear EOTV and the chemical systems, one slight change was made to the data just presented. By using the shuttle's nominal orbit altitude of 300 km as the initial altitude, the costs of the deployments decrease slightly because of the reduction in triptime by 1.5 days. The costs for the 12 engine system are :

First Roundtrip	:	$ 26,616,862
Middle Roundtrips	:	$ 10,083,300
Last Outbound Trip	:	$ 9,159,380

CHAPTER VII. OVERALL COST COMPARISON

Having determined the nuclear powered and solar powered systems which best satisfy the user's time and cost constraints, it is now time to compare these systems against each other and against the available chemical OTVs. As stated in the objectives, there will be 28 satellites to deploy (4 per year for 7 years).

For the electric systems, the following costs have been included in the analysis :

a) Purchase costs for the EOTVs.

b) Purchase costs for the replacement engines.

c) Earth to LEO launch costs for the GPS satellites, the EOTVs, and replacement engines and fuel.

d) Operations costs during the transfer orbits.

For the chemical systems, the costs included are the purchase costs for the upper stages and the cost to launch the payload and upper stage to LEO. The transfer times for the chemical systems are approximately six hours and the costs associated with tracking and guidance during the transfer orbits are negligible.

In this comparison, it was assumed that there are no failures of any of the systems. This is because there currently is no reliability data on any large scale electric system. The SERT II tests discussed in the literature review represent most of the testing for the type of electric engines considered in this analysis. There are no more

recent tests. The chemical systems being considered have not been used on enough missions to accurately determine their reliability.

ELECTRIC SYSTEMS

The only non-reusable electric system is the silicon array powered EOTV with 12 engines using mercury for the propellant. The per satellite deployment cost was calculated to be $ 23.34 million. This means it will cost $ 653.52 million to deploy 28 satellites.

Calculating the costs for the reusable systems is a bit more complex. The solar powered system uses Gallium Arsenide concentrators to power 12 mercury ion engines. Given an engine life of 15000 hours, a total of six EOTVs are needed. The first pair would handle the first ten satellites before needing to be replaced. Similarly, the second pair would also deploy ten satellites and the last pair would deploy only eight satellites. The costs to do this are :

6 initial roundtrips	:	$ 159,701,172
16 middle roundtrips	:	$ 161,332,800
6 last outbounds	:	$ 54,956,280
		$ 375,990,252

or $ 13,428,223 per satellite.

If the engine life is 20000 hours then only four EOTVs would be needed. Each pair of EOTVs would deploy 14 satellites. The costs would be :

4 initial roundtrips	:	$ 106,467,448
20 middle roundtrips	:	$ 201,666,000
4 last outbounds	:	$ 36,637,520
		$ 344,770,968

or $ 12,313,249 per satellite.

For the nuclear powered system, a 100 KW nuclear generator powering 37 Xenon engines, there is a slight difference in the calculations. Because the nuclear generator is designed to last for seven years and because it is expensive to purchase and deploy, it is not practical to scrap the whole EOTV when the engines burn out at two or three years. Therefore, at the end of the engines' useful life, they will be replaced. This means that the cost of replacement engines as well as the cost to bring them to LEO must be included in the total deployment cost. These calculations are shown in Appendix D with the following results :

15000 hour engines :	$ 573,450,000
20000 hour engines :	$ 537,630,000

CHEMICAL SYSTEMS

Since all the chemical systems considered (PAM D-II, IUS and CENTAUR-G) are non-reusable, the only costs to be considered are the purchase cost for the upper stage and the launch cost for the upper stage and the payload. The calculations for the launch costs are shown in Appendix H.

The purchase price for the PAM D-II varied from $ 6 Mil (74) to $ 10 Mil (78). Using the more optimistic price, the total deployment cost comes to $ 822.25 Mil or $ 29.366 Mil per satellite.

Using a purchase price of $ 84 Mil for the IUS and $ 30 Mil for the CENTAUR-G, the deployment costs are $ 3847.93 Mil and $ 2490.93 Mil respectively.

COST COMPARISON

Table 7.1 summarizes the costs to deploy all 28 satellites. It clearly shows that the reusable systems cost significantly less than the non-reusable chemical systems. For the 20000 hour engines, the solar EOTV costs only 41.9% as much as the best chemical system, the PAM D-II, and the nuclear EOTV only 65.4%. The fact that the nuclear EOTV was able to achieve such good results despite its being rather expensive to purchase and deploy, shows that the reusablity of a system can be very helpful in bringing down total deployment costs.

Comparing the two reusable EOTVs, the solar powered EOTV is better. It costs only 64% of the nuclear EOTV and results in a savings of $ 192.86 Mil over the total deployment. This is because the solar EOTV uses one third the number of engines, the power source weighs and costs less, and the launch costs are much lower. This agrees with the findings of Mr. R. M. Jones (8) which state that besides thruster efficiency, a low specific mass power supply is the most

important factor in electric propulsion.

Table 7.1. Total Deployment Costs ($ Million)

System	Total Cost	Cost per Satellite
Non-reusable Solar EOTV	653.52	23.34
Reusable Solar EOTV :		
15000 hour engines	375.99	13.43
20000 hour engines	344.77	12.31
Reusable Nuclear EOTV :		
15000 hour engines	573.45	20.48
20000 hour engines	537.63	19.20
Chemical Systems :		
PAM D-II	822.25	29.37
CENTAUR-G	2490.88	88.96
IUS	3847.93	137.43

CHAPTER VIII. SENSITIVITY ANALYSIS

SOLAR

The EOTV which gives the best cost performance is the reusable 1990's technology mercury fueled system with twelve engines (20000 hour life) and galium arsenide concentrator arrays placed at an initial inclination of 55 degrees. This system's performance is a result of two key assumptions:

1) That gallium arsenide concentrators with a degradation factor (R) of .05 and costing $225/watt will be available.

2) That the stated improvements in engine systems will occur.

It is important to see what effect changes in these assumptions has on the system performance. To accomplish this, the following cases were examined.

a) Ga-As concentrators with R = .1 and R = .2.

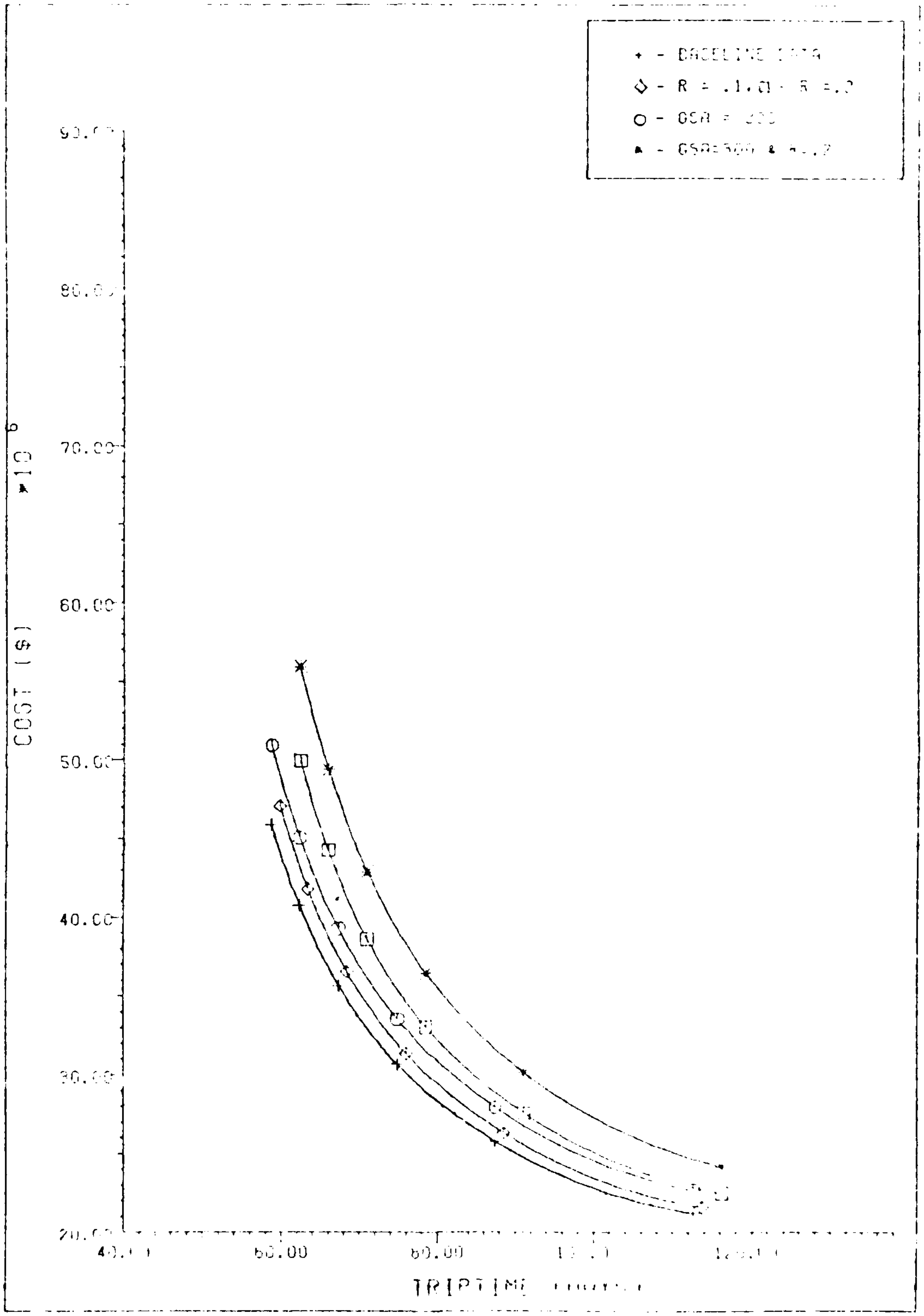
b) Ga-As concentrators with a cost of $300/W.

c) Ga-As concentrators with R = .2 and costing $300/W.

d) Present technology engines with the baseline Ga-As concentrator arrays.

Additionally, the effects from changes in the payload weight and in the operations costs were also examined.

The effects from changes in the degradation factor and costs of the solar arrays are shown in Figure 8-1. Increasing R to .1 produces very little change . The increase in solar array mass is only 20 kgs. This results in a 1.1 day increase in triptime. The cost for the initial



EFFECTS OF DEGRADATION & ... COST ON ...

Figure 3-1

roundtrip is increased by only $.39 million and for the total deployment, the increase is only $5.96 million or $.213 million per satellite.

For a degradation factor of .2, the changes are a little larger but still not significant enough to alter the results. The solar array mass is increased by 99 kg, triptime by 3.7 days, and overall costs by $21 million or $.75 million per satellite.

Increasing the solar array specific cost to $300/W does not affect the triptime of the EOTV. The only change is in the cost of the solar array which increases by $2.2 million per EOTV. This results in a total cost increase of $8.8 million or $.314 million per satellite.

For the worst case of R = .2 and solar array specific cost of $300/W, the increase in total cost is only $33.74 million or $1.205 million per satellite. The deployment costs for this system are still $160 million less than for the reusable nuclear system and $444 million less than for the PAM D-II.

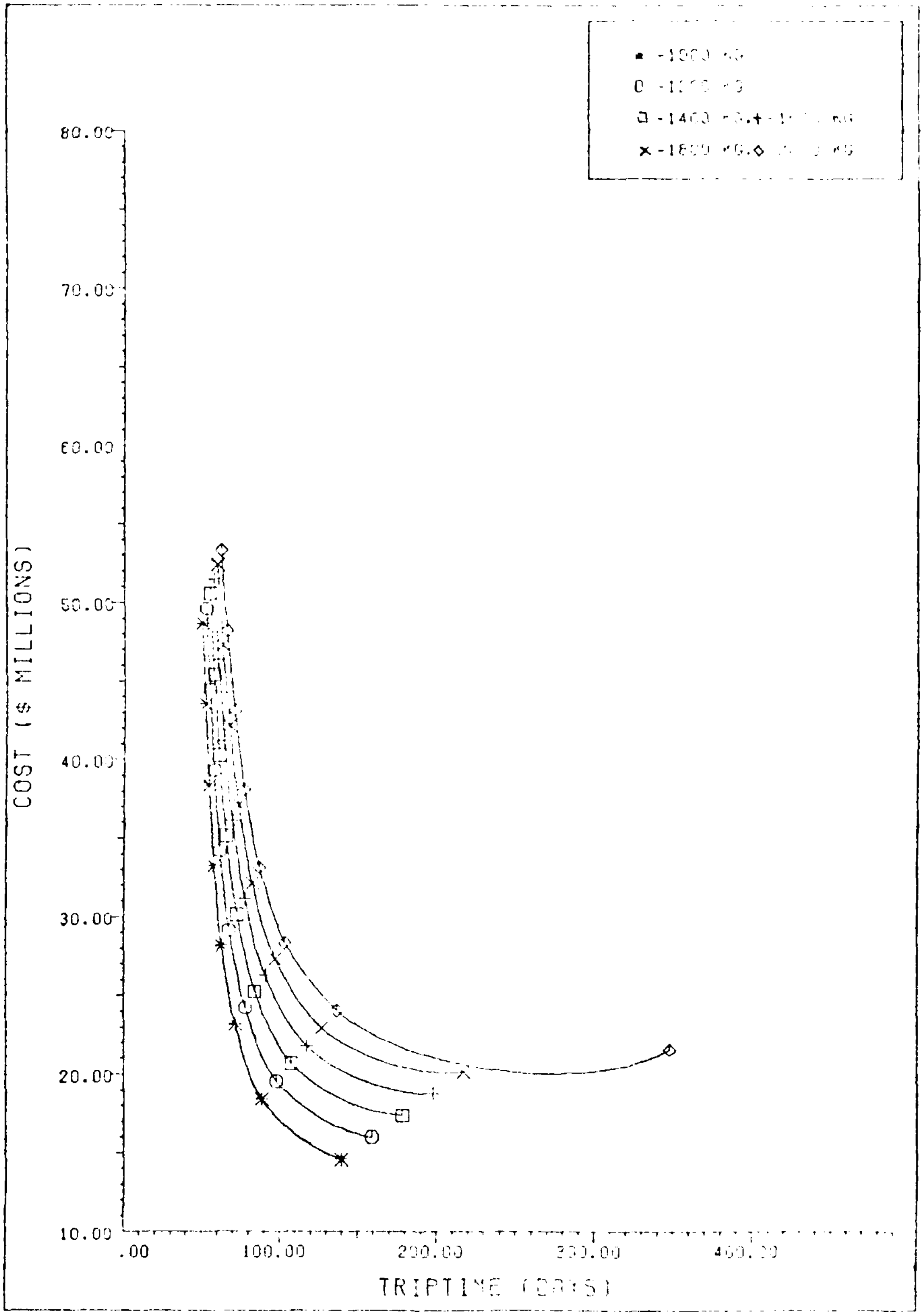
If nothing is done to improve the present engine systems and the only improvement in solar arrays is the developement of the Ga-As concentrator arrays, then 16 engines would be needed for the EOTV to have a triptime at or below 90 days. For this system, the outbound triptime is 86.2 days, the first roundtrip costs $42.67 million, the middle roundtrips cost $11.27 million each, and the last outbound trip costs $10.07 million. An EOTV could deploy a total of six

satellites and five EOTVs would be necessary for the deployment of 28 satellites. The total cost for this deployment would be \$466.5 million or \$16.66 million per satellite. This is still \$71 million less than the reusable nuclear EOTV and \$355.75 million less than the PAM-DII.

The effects of changing the payload weight are shown in Figure 8-2. Decreasing the payload weight decreases both the triptime and cost. In some cases it is possible to decrease the number of engines as well and realize a larger cost reduction. For the 1000 kg payload, eight engines still produce a triptime less than 90 days and the cost is \$5 million less than with 12 engines.

Increasing the payload causes both the triptime and cost to increase. While the magnitude of the increases are small, the triptimes increase enough so that they are no longer less than or equal to 90 days. In order to bring them back down to acceptable levels, extra engines are required. For the 2000 kg payload, 16 engines are needed for a triptime of 87 days. Each EOTV would still be able to deploy seven satellites and four EOTVs could handle the full deployment of 28 satellites. The cost for this would be \$430.3 million or \$15.37 million per satellite. This is only \$86 million more than for the baseline payload and is still much less than either the nuclear EOTV or the PAM D-II carrying the baseline payload of 1500 kg.

Changing the operations costs has little effect on the total cost of the deployment. Figure 8-3 shows that at the



TRIPTIME VS COST FOR VARIOUS PAYLOAD WEIGHTS

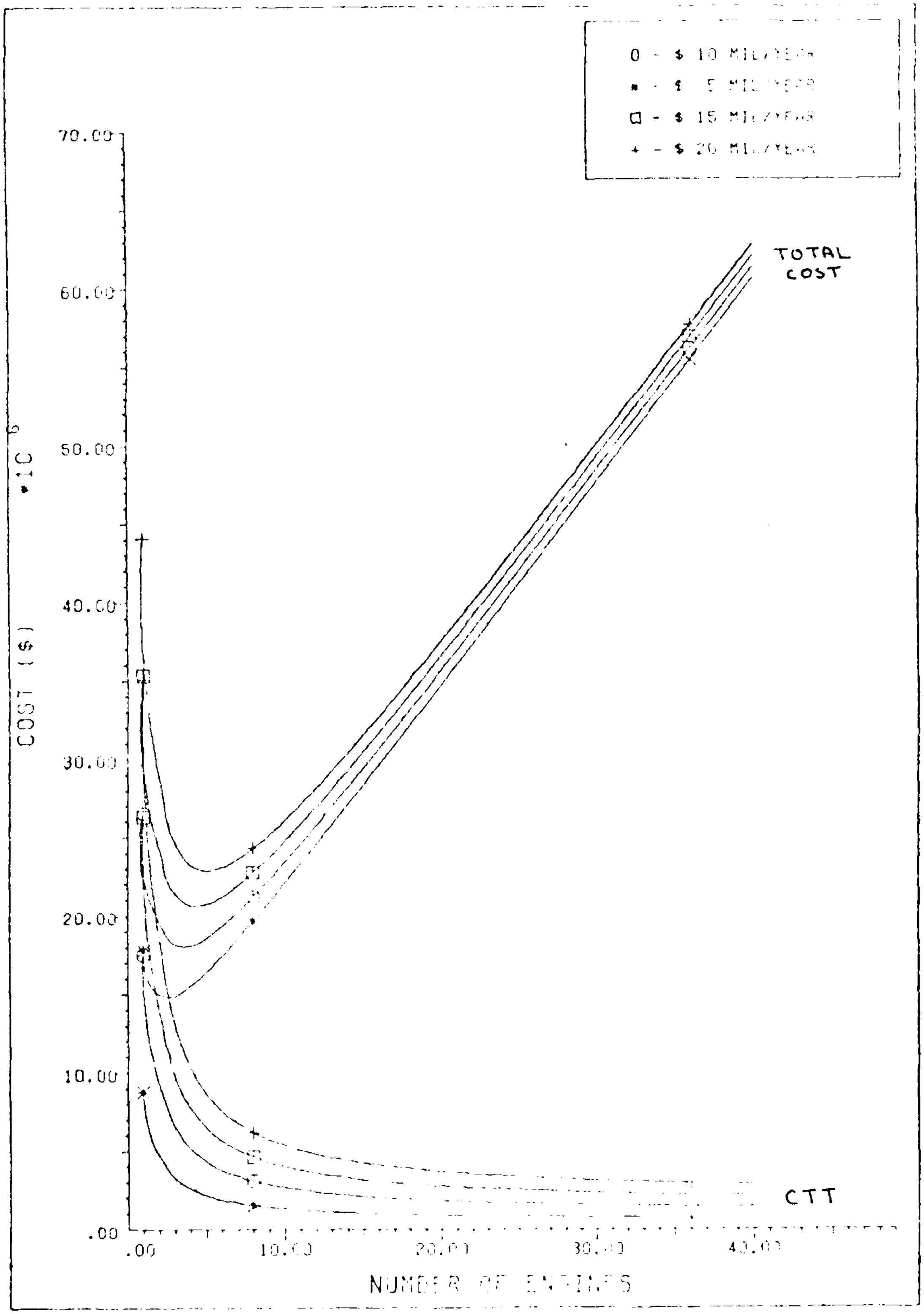
Figure 8-2

operating point of 12 engines, the difference in transfer time costs between the baseline operations costs of $10 million per year and $20 million per year is only $2.4 million. This means the increase in total deployment costs is only $67.2 million. The effect of changing operations costs would have been more significant if the operating point was at a lower engine number. Here the triptimes are greater and the operating costs constitute a larger percentage of the total cost.

The sensitivity analysis has shown that altering the assumptions made at the beginning of the study does not change the end results. The reusable solar powered EOTV is still the system which can deploy the 28 satellites for the least cost. While the number of engines needed varies from eight to sixteen depending on the payload weight, one fact never varied - this system can deploy the satellites for 60 % (or less) of the cost using the PAM D-II.

Table 8-1. Sensitivity Analysis Summary

	Total Cost ($ Million)
Baseline Systems :	
Reusable EOTV (R=.05, ASA=$ 225/W)	$ 344.77
PAM D-II	$ 822.25
Variations to baseline EOTV :	
Reusable EOTV (R=.2, ASA=$ 300/W)	$ 378.51
1984 tech. engines with Ga-As arrays	$ 466.50
2000 kg payload	$ 430.30
$20 Mil/year operations costs	$ 411.97



ENGINE NUMBER VS TOTAL COST FOR VARIOUS OPS COSTS

Figure 8-3

NUCLEAR

The sensitivity analysis performed on the nuclear OTV indicates that the power system is the most sensitive component. Analysis shows that the largest room for improvement lies in the reactor/heat exchanger. If the current 9 per cent efficiency could be doubled, a reduction in mass of approximately 500 Kg could be made by having less nuclear fuel and waste heat radiator. If the efficiency became 50 per cent, approximately 800 Kg could be saved. While this is a 10 per cent mass reduction, the overall results for the GPS mission scenario would not change. The SOTV mass is approximately 5000 Kg less than the NOTV. A much larger mass reduction would be necessary before the two became equitable.

Since the possibility of launching two or more satellite payloads at a time exists, the Xenon 100 Kw system was tested with two and three payloads per mission. The results indicate that because the power plant is so massive, the additional satellite mass represents only a small increase. The effect of additional payloads is shown below. The

XE, 1995, delinc = 0

	# of GPS Payloads		
	1	2	3
Trip time out (days)	70.41	86.46	102.50
Total cost (millions $)	79.84	80.34	80.83

results indicate that if the mission scenario allowed

multiple payload launches, the NOTV would be much closer to the SOTV in terms of performance. In terms of cost however, the SOTV would retain its lead, while the chemical systems would be pushed even farther behind.

Finally, it is felt that improvements in the power plant specific mass would greatly increase the performance and decrease the cost characteristics for the nuclear OTV. For instance, using the reactor in the one megawatt power range, the power plant specific mass is reduced from 30 Kg/Kw (100 Kw) to 12 Kg/Kw. At the 100 Kw level, a reduction of 20 per cent to 24 Kg/Kw produces the following results for the initial purchase, launch, and mission costs. Once again,

Xenon, 1995, 1500 Kg payload

	100 Kw (30 Kg/Kw)	100 Kw (24 Kg/Kw)
Total cost (millions $)	79.84	75.45

this would not change the overall results for this particular mission.

A nuclear OTV is extremely massive. It is sensitive to very little except that which changes its power plant specific mass. The only factor which shows opportunity for serious improvement is the reactor/heat exchanger/radiator efficiency. A reduction here would decrease the overall power plant specific mass, mission trip time, and total costs. These reductions would not change the overall results for this mission. However, for other missions, these reductions could have a large impact.

CHAPTER IX. CONCLUSION AND RECOMMENDATIONS

SUMMARY

The overall objective of this research was to determine the feasibility of and the cost optimum system for using electric orbit transfer vehicles to move Block 3 GPS satellites from LEO to a 10,900 nautical mile orbit and to compare it with chemical OTVs.

For the EOTV, the propulsion systems considered were present and 1990's technology ion engines using mercury, xenon or argon for a propellant. There were two power sources evaluated, a nuclear reactor and solar arrays. The nuclear reactor used was the 100 KW reactor being developed by the SP-100 program. Only flat-plate silicon cell arrays and gallium arsenide concentrator arrays were examined as possible solar power sources.

Because the problems and constraints vary depending on the power source used, two separate but similar methodologies were used in this thesis. A system cost model which combines payload, power source, trajectory, and earth-to-LEO launch parameters with algorithms characterizing the electric propulsion system was used. The model produced a set of costs for each system considered. The goal was to find the least costly nuclear powered and solar powered systems which had a triptime equal to or less than 90 days.

These systems were then compared with three chemical upper stages: PAM D-II, CENTAUR-G, and IUS. The basis for

the comparison was the total cost to deploy 28 GPS satellites at a rate of four per year for seven years.

CONCLUSION

The results indicate that the best overall system for deploying GPS satellites is the reusable solar powered electric OTV. This system has an outbound triptime of 86 days and is the least expensive. The table below summarizes the individual systems costs (in millions of $) for deploying 28 GPS satellites.

System	Cost
Reusable Solar EOTV (20000 hr engines)	344.77
Reusable Nuclear EOTV (20000 hr engines)	537.63
Non-reusable Solar EOTV	653.52
PAM D-II	822.25
CENTAUR-G	2490.88
IUS	3847.93

The sensitivity studies showed that changes in the degradation factor and the costs of the solar arrays produced only small increases in the total deployment costs. They did show that an EOTV using present technology engines and 1990's technology Ga-As concentrator arrays could still deploy the satellites for 57% of the cost using the PAM D-II. This means that the only technology barrier in the way of the developement of an EOTV is the completion of the developement of the Ga-As concentrators presently under way.

In evaluating the power sources, several important findings surfaced. For the nuclear reactor, it was noted that for a GPS payload weight, the reactor is at the low end of its operating range. In this region the powerplant specific mass is 30 kg/KW. For efficient use of the reactor, operation in the 1 MW region is desirable. This reduces the specific mass to 12 kg/KW. For very massive payloads (thousands of kilograms), the nuclear reactor becomes an attractive power source. Another problem with the nuclear powered OTV is that large waste heat radiators are necessary because of the reactor's poor efficiency (9%). More efficient reactors and heat radiators are needed to help reduce the power system mass.

In the evaluation of solar power sources, it was found that flat plate silicon cell arrays are unacceptable for use on a reusable EOTV. Their high degradation factor necessitates the use of extremely large arrays. For an array to be acceptable, its end-of-life degradation factor must be in the range of .2 or lower. This is illustrated by the performance of the EOTV utilizing gallium arsenide concentrators.

RECOMMENDATIONS

As a result of this thesis effort, several recommendations and areas for further study present themselves. The authors feel that the results indicate such a great potential for cost savings that more detailed cost

and engineering studies of these systems are warranted. Also, performing a similar study with the assumption of the availability of a space station would most likely produce further support for the use of EOTVs.

In this thesis, only two types of solar arrays were considered. There are other types of arrays as well as different thickness of cells and cell covers that should be evaluated for use on EOTVs. The developement of a set of guidelines to aid a designer in choosing the proper array type and cell and cover thicknesses would be a worthwhile undertaking.

A more detailed analysis of the tradeoffs between solar cell type, orbit trajectory and solar cell degradation would also aid in the design of an EOTV.

An analysis of orbit transfer trajectories to allow for the deployment of multiple payloads into similar or different orbits is necessary to define the delta V requirements of an OTV.

Further studies using other light to medium weight payloads and various orbits would define the missions for which the different power sources are best suited. In such an analysis, the size of the nuclear power source should be varied in order to assess its true potential.

With the present economic problems faced by the nation resulting in smaller and tighter budgets for the various agencies involved with the space efforts, it is essential that steps be taken to make their operations more economical.

Such a step would be the use of reusable electric orbit transfer vehicles as a means of transfering those payloads which are not time critical in their deployment. Not only will the cost savings over the presently used chemical OTVs be considerable, the technical knowledge gained will further aid in man's exploration of space.

APPENDIX A: BASIC PROGRAM SAMPLE LISTING

```
LIST
100 OPEN "hq4" FOR OUTPUT AS #1
200 OPEN "hq5" FOR OUTPUT AS #2
300 OPEN "hq6" FOR OUTPUT AS #3
400 REM  ************************
500 REM  engine parameters
600 REM  ************************
700 NSUBT=.922
800 NSUBF=.9
900 ISP=2900
1000 THRUST=.129
1100 FFLOW=THRUST/(ISP*9.8)
1200 ENGPOW=THRUST*ISP*.0098/(2*NSUBT*NSUBF)
1300 ENGM=40
1400 REM  ************************
1500 REM   orbit parameters
1600 REM  ************************
1700 MUERTH=398603.2
1800 RERTH=6378.165
1900 ORBINC=28.5
2000 RAD1=300
2100 RAD2=20186.81
2200 DELINC=(55-ORBINC)/57.3
2300 RADIU1=RAD1+RERTH
2400 RADIU2=RAD2+RERTH
2500 VSUBI=SQR(MUERTH/RADIU1)
2600 VSUBF=SQR(MUERTH/RADIU2)
2700 DELV=SQR((1-(2*(VSUBF/VSUBI)*COS(1.414*DELINC))+
                                        (VSUBF/VSUBI)^2))*1000
2800 REM  ************************
2900 REM  solution equations
3000 REM  ************************
3100 LENGTH=30
3200 SATC=75000000#
3300 DRAG=.0001
3400 STRM=150
3500 GNCM=50
3600 GNCC=1000000!
3700 SATM=1500
3800 SUMT=1000000000#
3900 FOR I=1 TO 43 STEP 1
4000 FOR J=1 TO 30
4100 NUMENG=I
4200 TENGM=ENGM*NUMENG
4300 POWER=ENGPOW*NUMENG
4400 POWERM=1600+(11.3*POWER)
4500 POWERC=1[illegible]*POWERM
4600 STRCST=(1000*STRM)
```

```
4700 LMASS=STRM+TENGM+FUELM+SATM+POWERM+GNCM
4800 TMASS=(1.1[illegible]*LMASS)
4900 TTO=((1+I[illegible])*LMASS*VSUBI*DELV/(NUMENG*THRUST))
5000 TTB=((1-I[illegible])*(LMASS-SATM)*VSUBI*DELV/(NUMENG*THRUST))
5100 TT=(TTO+TTB)
5200 FUELM=([illegible]*TT)*NUMENG
5300 FUELC=([illegible]*FUELM)
5400 TENGC=(NUM[illegible]*[illegible]000!)+FUELC
5500 LAUNCC=((LENG[illegible])/.75)*65000000#
5600 OPSCST=(.31[illegible]TT)
5700 HWRCST=[illegible]+TENGC+GNCC
5800 TCST=LAUNCC+OPSCST+HWRCST
5900 NEXT J
6000 IF TCST > SUMT THEN GOTO 6500
6100 SUMT=TCST
6200 BEST=I
6300 BESTT=TT/86400!
6400 BESTC=TCST
6500 PRINT #1,NUMENG,;",":TTO/86400!;",":TTB/86400!;",";TT/86400!
6600 PRINT #2,NUMENG;",";LAUNCC;",";OPSCST;",";HWRCST;",";TCST
6700 PRINT #3,TT/86400!;",";TCST
6800 NEXT I
6900 PRINT BEST,BESTT,BESTC
7000 END
Ok
```

APPENDIX B: OTV SYSTEM DESIGN

The full design of an nuclear electric OTV was not considered a part of this thesis effort. However, several proposed systems came to the authors attention, and this paper assumes the existance of a modification of these. Therefore, it is felt that an explanation of the proposed system is in order. This explanation will be approached by explaining the OTV in terms of the same divisions which were used in the methodology section. These, to remind the reader, are:

PROPULSION SYSTEM	POWER SYSTEM	OTHER STRUCTURES
Engines	Reactor	Boom
Power processors	Radiator & tubes	Van Allen Belt protection
Fuel and tanks	Thermoelectric Devices	Satellite-Shuttle adapter
Associated strutural hardware	Shielding	
	Pumps, working fluid	

The overall system design of the OTV is presented below.

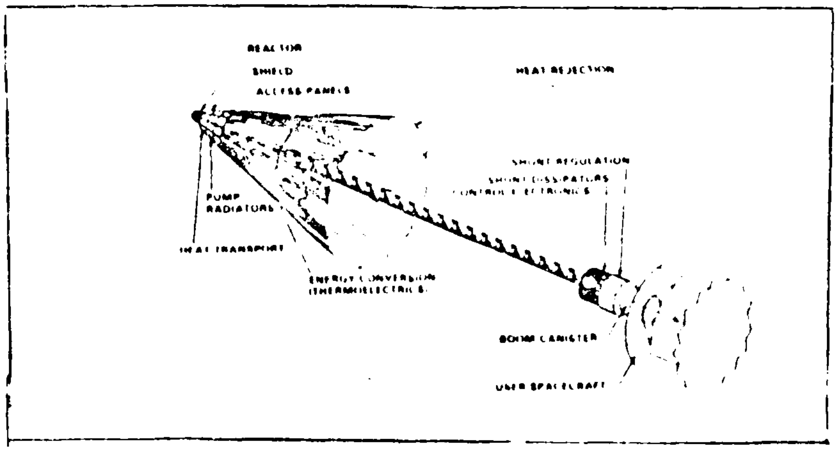


Figure B-1. Suggested Nuclear OTV Design (15)

Approaching the system from the reactor end, we will cover

the power system first, then the engines, and finally any other structures.

SP-100 PROGRAM

While there are a few arguments against using a NPS in outer space (15, 16, 20, 61), NASA and the DOD have already decided to develop a space qualified NPS under the auspices of the SP-100 program (77). This reactor, which is the model used for this research, has the design objectives listed below:

SP-100 GOALS (Buden)

Performance

Power output, net to user (KW)	100
Output variable up to 100 Kw	
Full power operation (years)	7
System life (years)	10
Reliability (%)	
1st system, 2 years	.95
2nd system, 7 years	.95
Multiple restarts	

Physical constraints

Mass (Kg)	3000
Size, length within the STS envelope (m)	6.1

Interfaces

Reactor induced radiation after 7 yr operation, 25 m from forward end of reactor	
Neutron fluence (n/cm2)	1013
Gamma dose (rads)	5x105
Mechanical	STS launch conditions
Safety	Nuclear Safety Criteria and Specifications for Space Nuclear Reactors

Currently there are three choices in contention for use as a nuclear power source in space. These differ mainly in the

manner in which they convert the reactor thermal power to electricity. They include thermionic, thermoelectric, and stirling conversion. The method and system used for this analysis is the thermoelectric.

A picture of the reactor used for this system is shown in Figure B-2, with a mass to power graph depicting the mass of the total power system in Figure B-3.

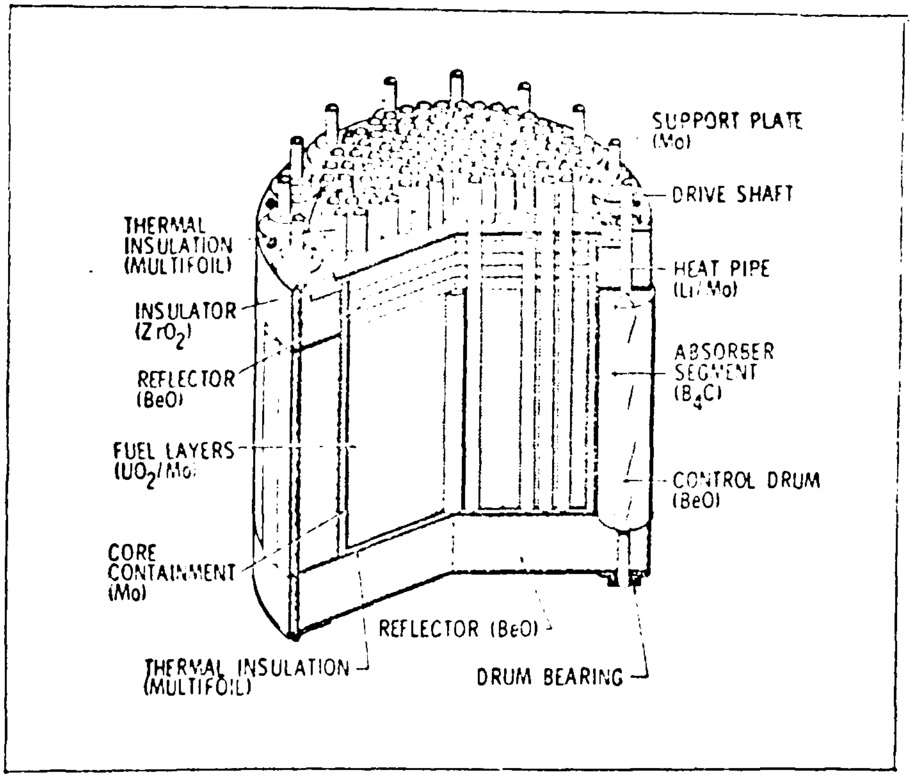


Figure B-2. A Typical Space Nuclear Reactor (56)

Behind this reactor is the shielding which helps to shield the OTV components from the radiation hazards of the reactor. This is one place where the engines can be placed, located axially around the longtudinal center of the OTV. Immediately behind this radiation shield lies the largest

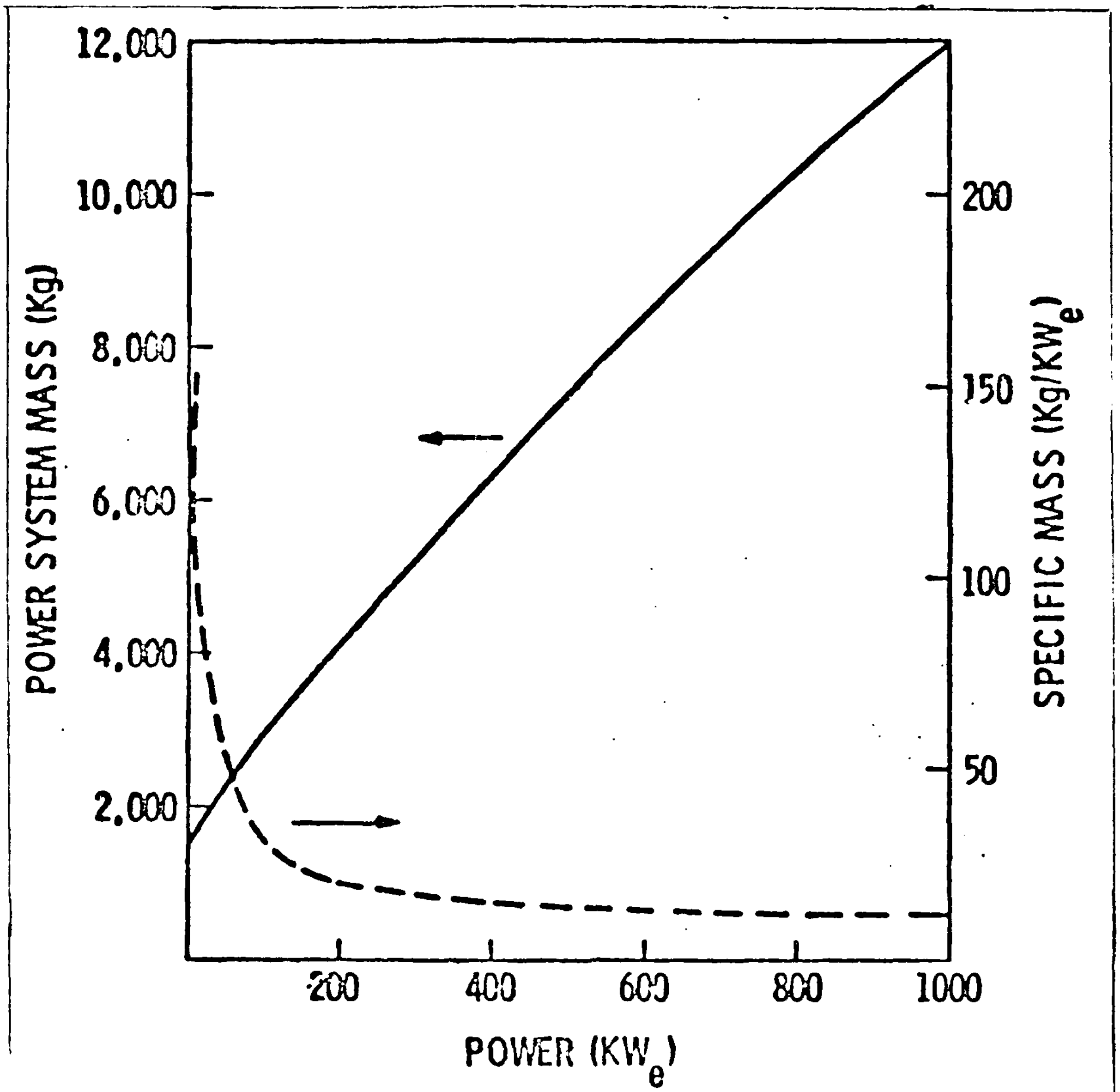


Figure B-3. Nuclear OTV Power Plant Mass (15)

part of the power system, the waste heat radiator. Within this radiator lies the thermoelectric converters.

The reactor produces energy in the form of heat which is transfered to the working fluid which flows through the pipes in the reactor. This heat is transfered, through pipes, to the thermoelectric converters where it is converted to electricity. The waste heat from these converters is transported to the radiator to be radiated into space. The working fluid then returns to the reactor to be reheated.

The thermoelectric converters operate by changing some of the heat of the working fluid into electricity, thereby reducing somewhat the temperature of the fluid. The radiator reduces the heat of the fluid to its normal operating temperature prior to returning it to the reactor.

PROPULSION SYSTEM

The propulsion system is composed of the ion engines, their fuel and tanks, the power processors which condition the electricity, and any other associated structures which these components require. A picture of an ion engine is presented below. Its operation will not be explained here, as it can be found in any good propulsion book.

The power processors prepare the raw electricity for use by the engines, and make up most of the cost of the engine system (66). The fuel costs, as compared to the overall propulsion system, are very small. The engines are expected to last for 15,000 to 20,000 hours, due to the internal grid.

If this grid can be removed, then the projected lifetime might exceed 30,000 hours (54, 66).

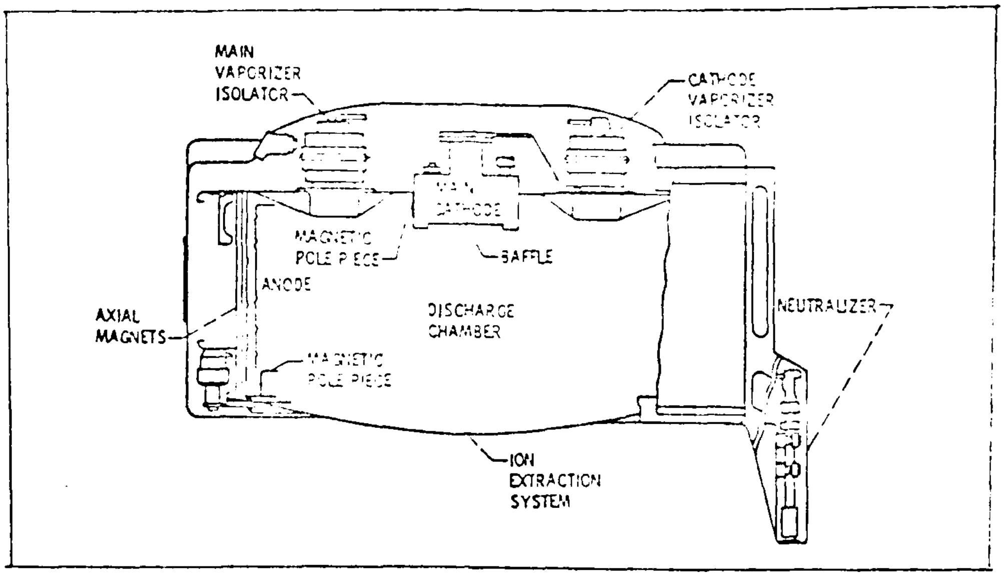


Figure B-4. Typical Ion Engine Design (52)

OTHER STRUCTURES

Other structures include the 25 meter boom to connect the satellite or payload, the shuttle adapter for the OTV, any extra radiation protection, and the guidance, navigation, and control system. The purpose of the boom is to remove the radiation sensitive payload from the influence of the radioactive reactor. In particular, for this model the boom telescopes out from the inside of the radiator. This allows the entire boom and payload to be placed within the radiator initially, then expanding to its full length after deployment from the shuttle. This keeps the launch costs low as the entire system can be launched using only half of the shuttle payload bay. Once in space, the boom expands at a very slow

rate. Since the reactor requires 30 minutes to warm up, time is not critical. The feasibility of using an expanding boom was discussed with several experts (17, 66) and found to be a realistic approach.

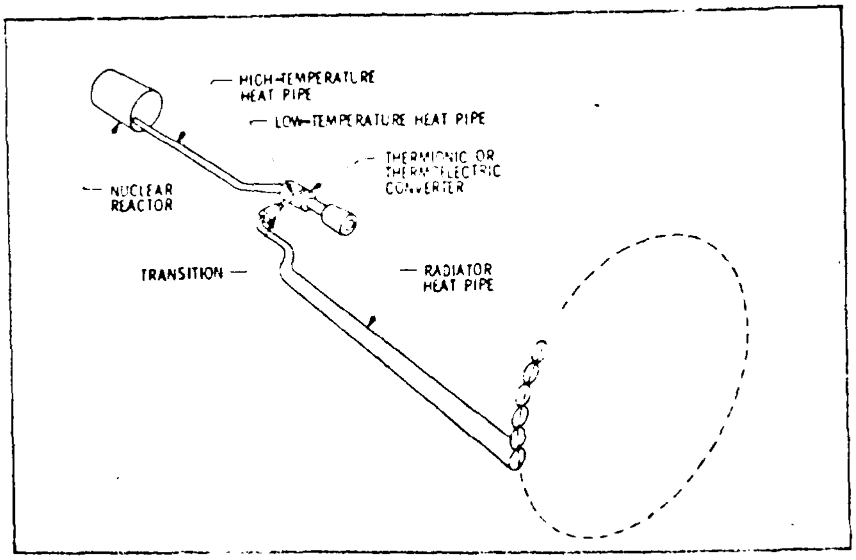


Figure B-5. Pipe Heat Rejection Concept (56)

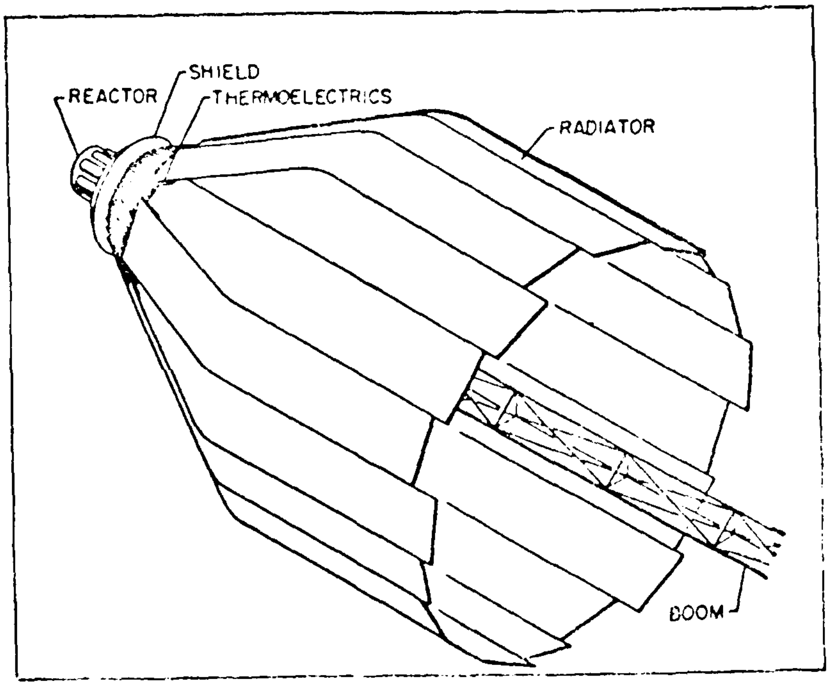


Figure B-6. Plate Heat Rejection Concept (56)

APPENDIX C: EQUATIONS USED

The description and explanation of the equations used in the Basic program analysis are presented below. The variables will be listed first, and those that are constants will have their value after them. Then the equations will presented by OTV design section, leading to the final cost equations.

VARIABLES

Gravitation	g (9.8 m/sec2)
Thruster efficiency	NSUBT
Power processor efficiency	NSUBP (.9)
Specific impulse	ISP
Thrust	T (.129 N)
Engine mass	ENGM
Power required by the engine	ENGPOW
Mu for earth	MUERTH (398603.2
Radius earth	RERTH (6378.165 Km)
Initial orbit inclination	ORBINC
Initial orbit altitude	RAD1
Final orbit altitude	RAD2 (20186.81 Km)
Final minus Initial orbit inclins	DELINC
Initial orbit radius	RADIU1
Final orbit radius	RADIU2
Initial orbit velocity	VSUBI
Final orbit velocity	VSUBF
Size of OTV	LENGTH (30 ft)
Mass of satellite	SATM (1500 Kg)
Cost of satellite	SATC (75000000 $)
Drag factor	DRAG (.001)
Structure mass	STRM (150 Kg)
Guidance, Navigation and Contros	GNCM (50 Kg)
Guidance, Navigation and Contrmisc	GNCC (1000000 $)
Number of engines on OTV	NUMENG
Power required for OTV	POWER
Mass of power supply	POWERM
Cost of Power supply	POWERC
Structure cost	STRCST (150000 $)
Launch mass	LMASS
Total mass	TMASS
Trip time out	TTO
Trip time back	TTB
Total trip time	TT
Fuel mass	FUELM
Fuel cost	FUELC (15 $/Kg)
Total engine system cost	TENGC

Launch cost	LAUNCC
Operations cost	OPSCST
Hardware cost	HWRCST
Total cost	TOTCST

ENGINE EQUATIONS

The first set of equations (42) concerns the engine systems:

$$\text{Fuel flow} = T / (Isp * g) \tag{C-1}$$

$$\text{Engine power} = T * Isp * g / (1000 * 2 * NSUBT * NSUBP) \tag{C-2}$$

These two numbers, combined with the input parameters, give all of the required engine parameters necessary for the analysis.

The equations dealing with the orbit determinations and required changes in velocity are (7):

$$\text{Initial orbit velocity} = (MUERTH/RADIU1)^{.5} \tag{C-3}$$

$$\text{Final orbit velocity} = (MUERTH/RADIU2)^{.5} \tag{C-4}$$

The orbit radius is simply the orbit altitude plus the radius of the earth.

The power, trip time, and associated equations are the important equations in that the entire analysis was performed in terms of costs per mass, time, and launch. These equations involve a closer look and greater discussion. We will skip the simple equations such as total engine mass equaling the number of engines times the individual engine mass, and proceed with the ones which are not so easy to determine. The first is

$$\text{Power mass} = 1600 + (12.5 * \text{Power}). \quad \text{(C-5)}$$

This is derived from the mass to power chart shown in Figure B-3. It was linearized over the range of interest (up to 100 Kilowatts), with the above parameters resulting. The power cost

$$\text{Power cost} = 13333 * \text{POWERM} \quad \text{(C-6)}$$

results from the estimated power plant cost (42) divided over the expected mass of 3000 Kg.

The trip time equation was discussed in the methodology and will not be repeated here. The only difference between the trip time out and trip time back was the effect of the change in the satellite mass, which would be left at the higher orbit.

The fuel equations used were:

$$\text{Fuel mass} = (\text{FFLOW} * \text{TT}) * \text{NUMENG} \quad \text{(C-7)}$$

$$\text{Fuel cost} = 15 * \text{FUELM} \quad \text{(C-8)}$$

As stated previously, the constant fuel cost was assumed due to lack of other figures for two of the fuels. Recent research has shown that this assumption is valid unless an enormous amout of Xenon (100 metric tons) is required (43).

The rest of the equations used in the analysis are either self explanatory or have been explained previously. Most of them are extremely simple in nature and a quick look at the computer program listing (Appendix A) will explain them.

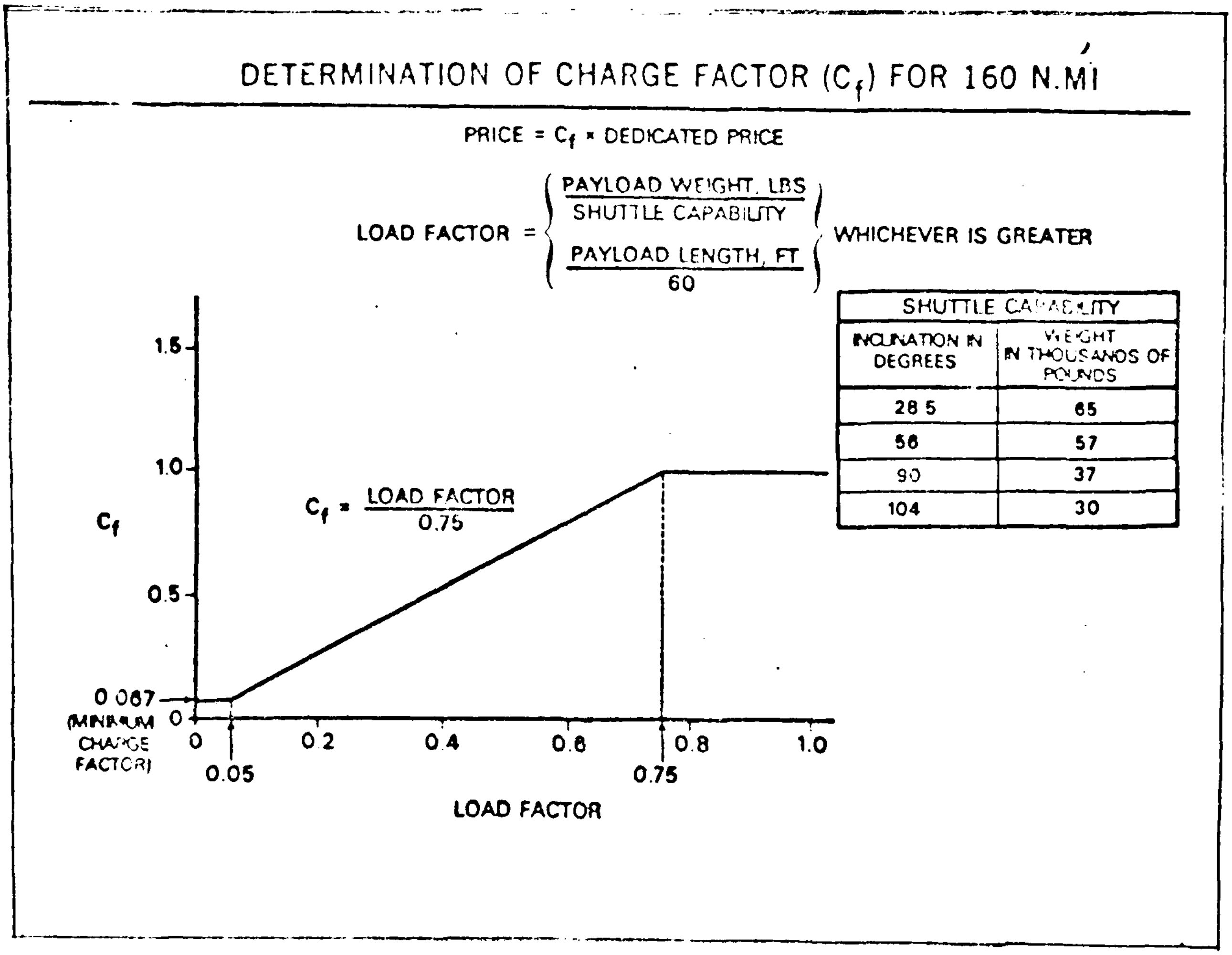


Figure C-1. Shuttle Launch Cost Chart (75)

APPENDIX D: COST CALCULATIONS (In millions of $)

HG, 1995, 43 ENGINES, DELINC = 0, 300 KM INITIAL LAUNCH
15000 HR ENGINE LIFE

Cost of procurement, launch, first trip	162.90
Fuel costs for 27 more trips (86,778 Kg)	1.30
Cost to move fuel to orbit	290.88
Cost to replace engines (43 engines, 4 times)	60.20
Cost to move engines to orbit	23.06
Ops cost for 27 missions	89.37
Total	627.71

HG, 1995, 43 ENGINES, DELINC = 0, 300 KM INITIAL LAUNCH
20000 HR ENGINE LIFE

Cost of procurement, launch, first trip	162.90
Fuel costs for 27 more trips	1.30
Cost to move fuel to orbit	290.88
Cost to replace engines (43 engines, 4 times)	60.20
Cost to move engines to orbit	23.06
Ops cost for 27 missions	89.37
Total	627.71

XE, 1995, 37 ENGINES, DELINC = 0, 300 KM INITIAL LAUNCH
15000 HR ENGINE LIFE

Cost of procurement, launch, first trip	159.68
Fuel costs for 27 more trips (62,991 Kg)	0.94
Cost to move fuel to orbit	211.14
Cost to replace engines (37 engines, 6 times)	77.70
Cost to move engines to orbit	29.76
Ops cost for 27 missions	94.23
Total	573.45

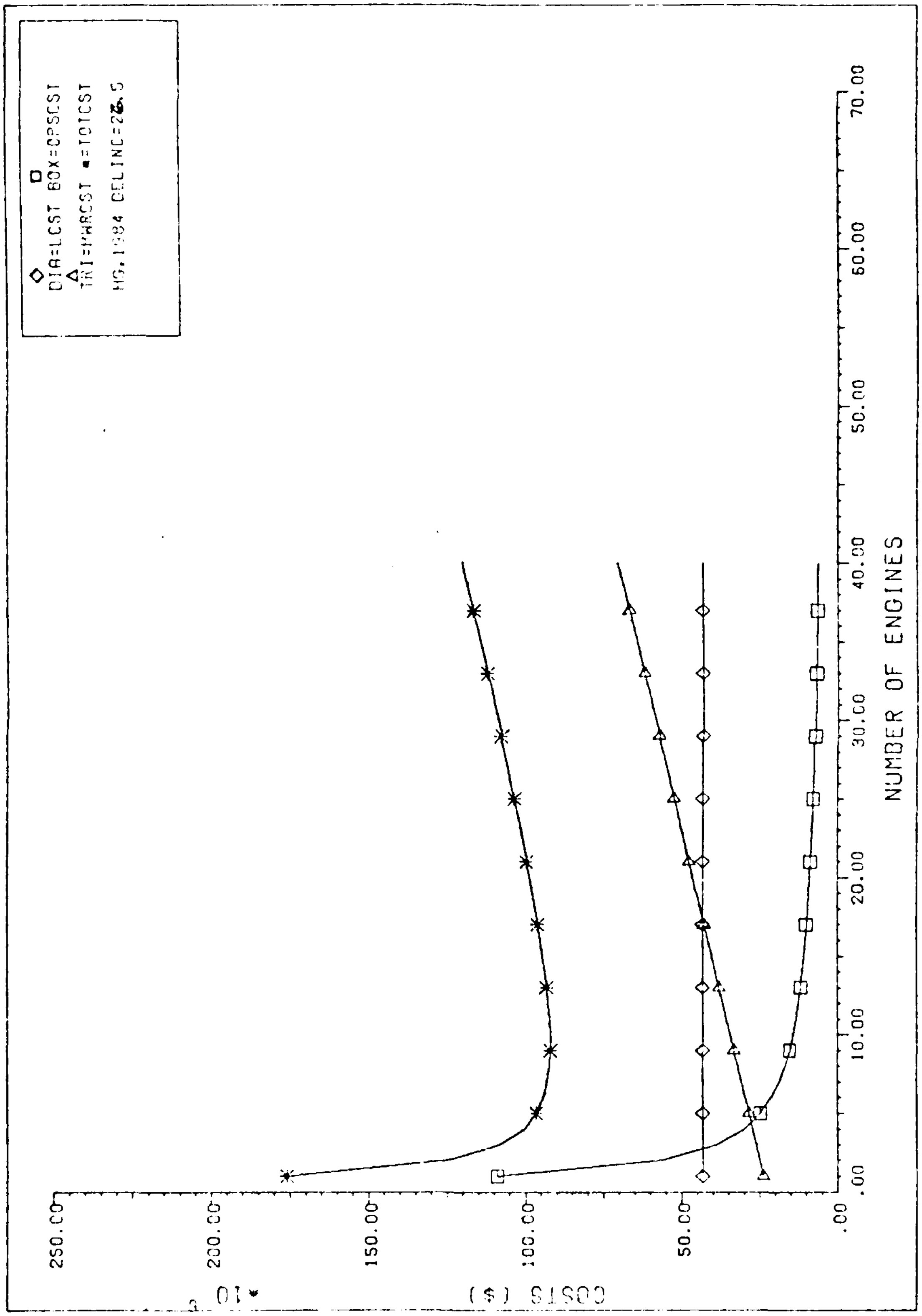
XE, 1995, 37 ENGINES, DELINC = 0, 300 KM INITIAL LAUNCH
20000 HR ENGINE LIFE

Cost of procurement, launch, first trip	159.68
Fuel costs for 27 more trips	0.94
Cost to move fuel to orbit	211.14
Cost to replace engines (37 engines, 4 times)	51.80
Cost to move engines to orbit	19.84
Ops cost for 27 missions	94.23
Total	537.63

	HG		XE	
Engine life	15000	20000	15000	20000
# of trips before engines fail	5.19	6.92	4.92	6.56
Rounded off	5	6	4	6
# of engine changes required	1.8	1.3	2.5	1.3
Rounded off	2	2	3	2
Total number of engine changes required (2 OTV's)	4	4	6	4

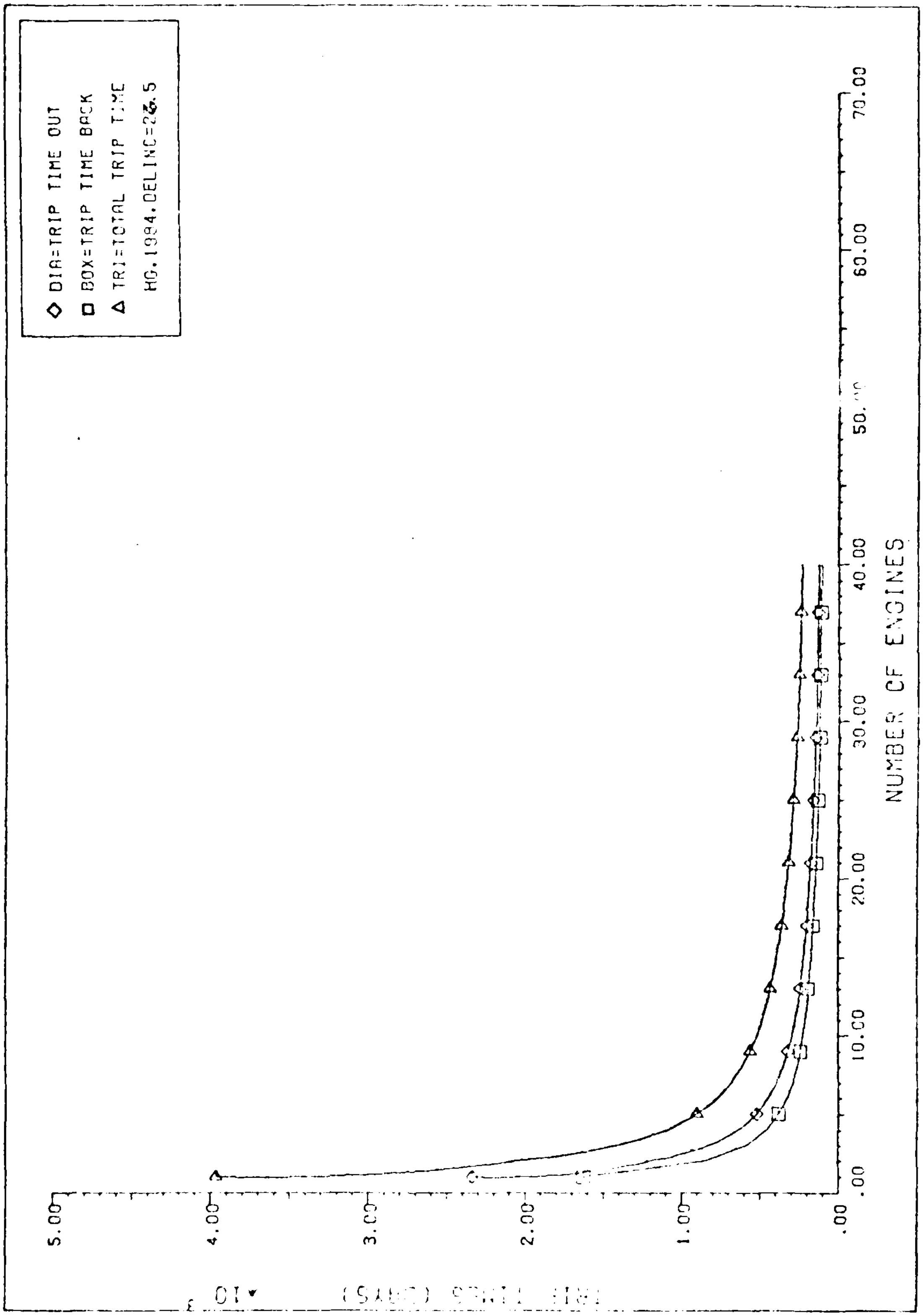
Cost per engine ($)	350,000
HG (43 engines)	15.05 million
XE (37 engines)	12.95 million
Mass of engines (HG, XE)	40 Kg
Fuel per round trip	
HG	3,214 Kg
XE	2,333 Kg
Ops cost per trip	
HG	3.30 million
XE	3.48 million
Max shuttle capacity 55 degree inclination	25,855 Kg

Appendix E: Supplemental Graphs - Nuclear



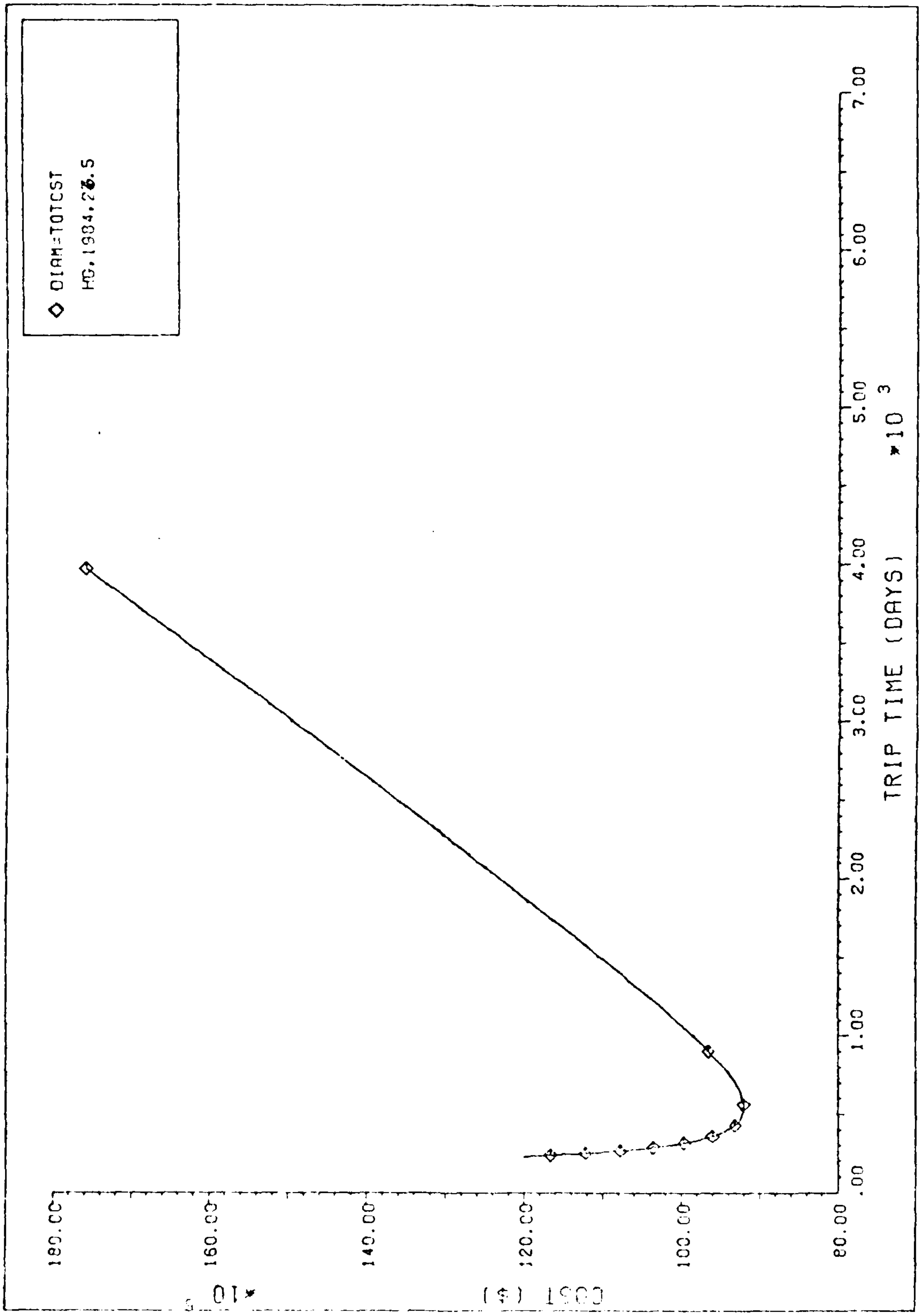
COSTS VS. NUMBER OF ENGINES HG.1984.26.5

Figure E-1



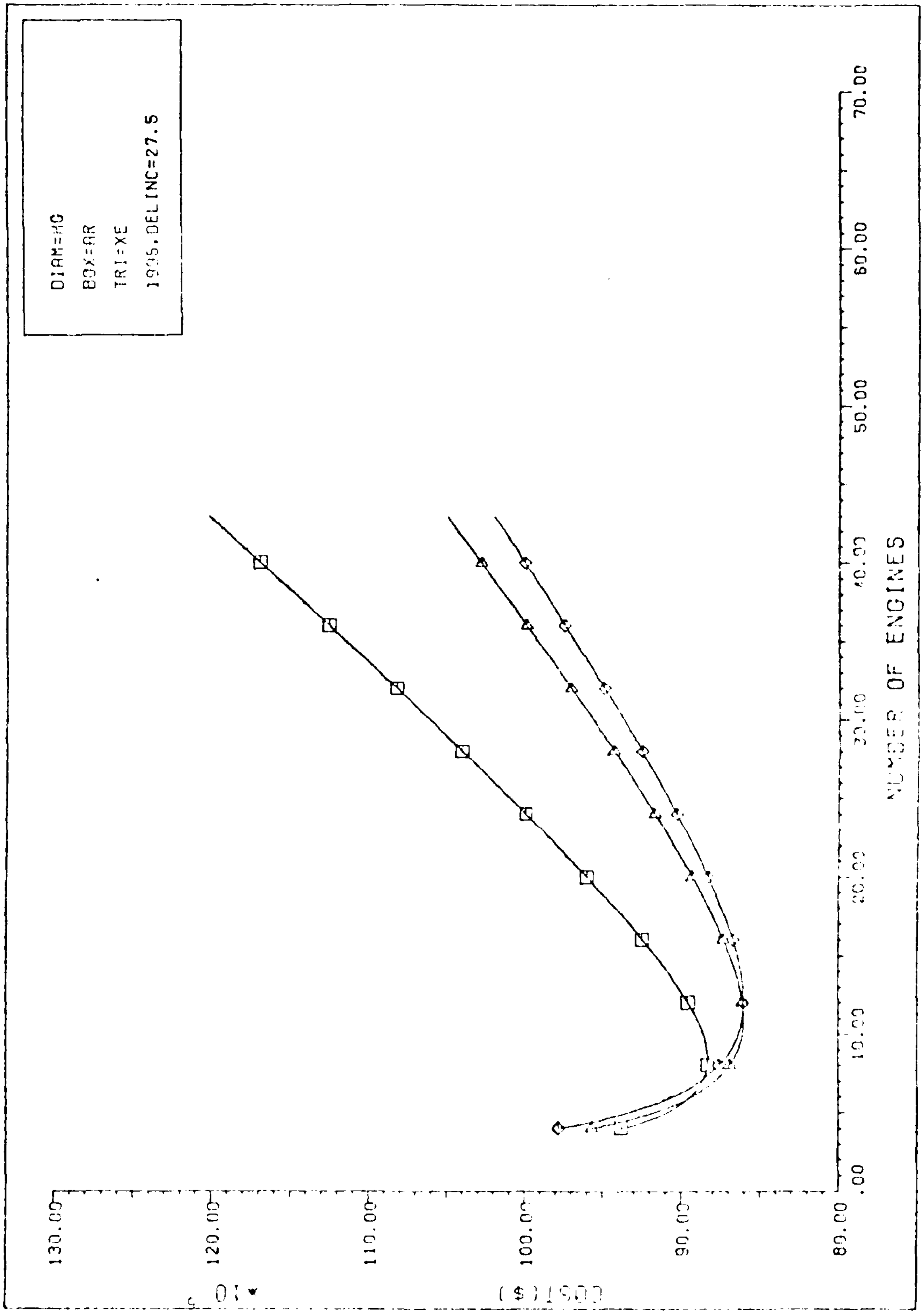
TRIP TIMES VS. NUMBER OF ENGINES HG.1994.26.5

Figure E-2



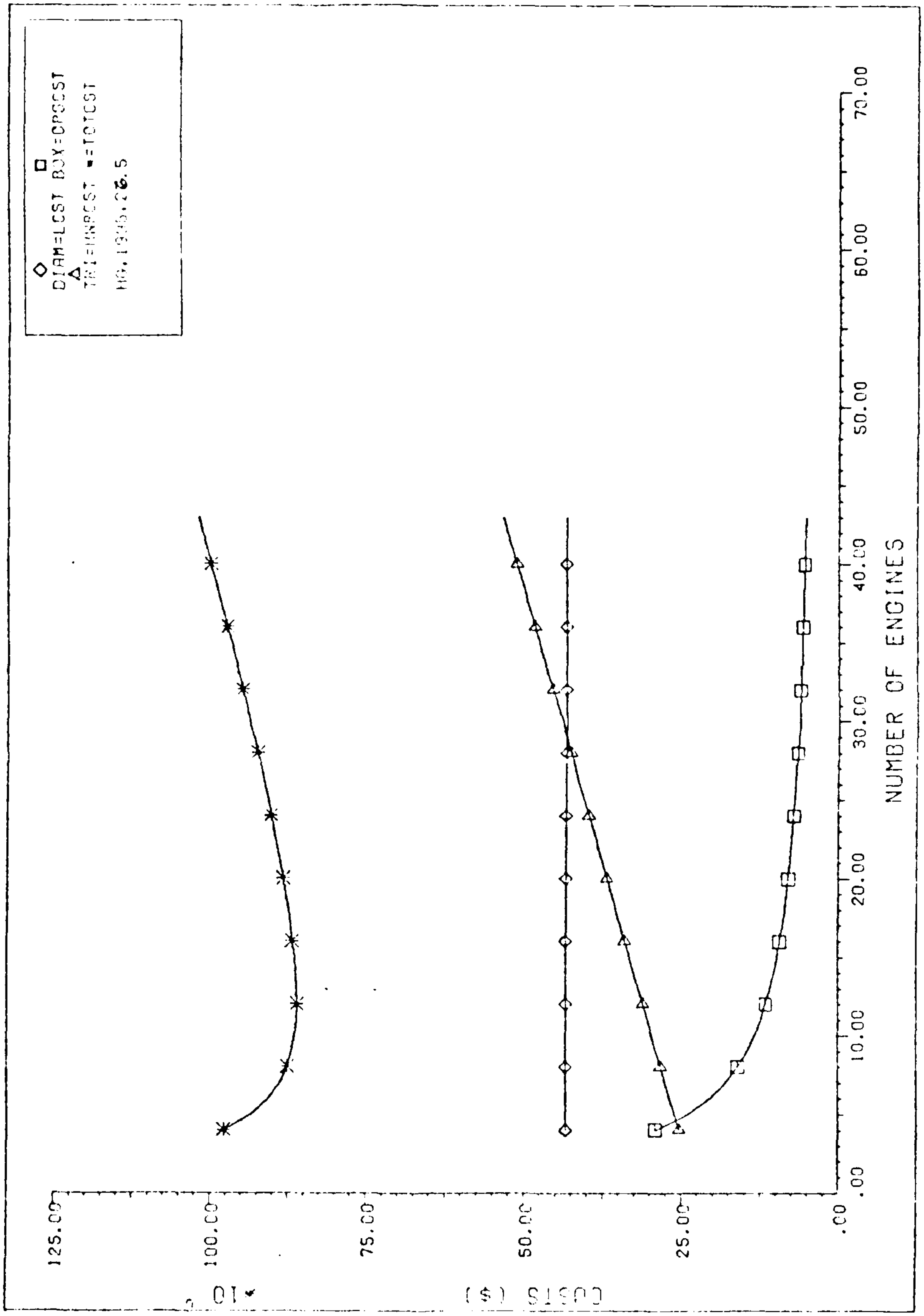
TOTAL COST VS. TRIP TIME HG.1984.26.5

Figure E-3



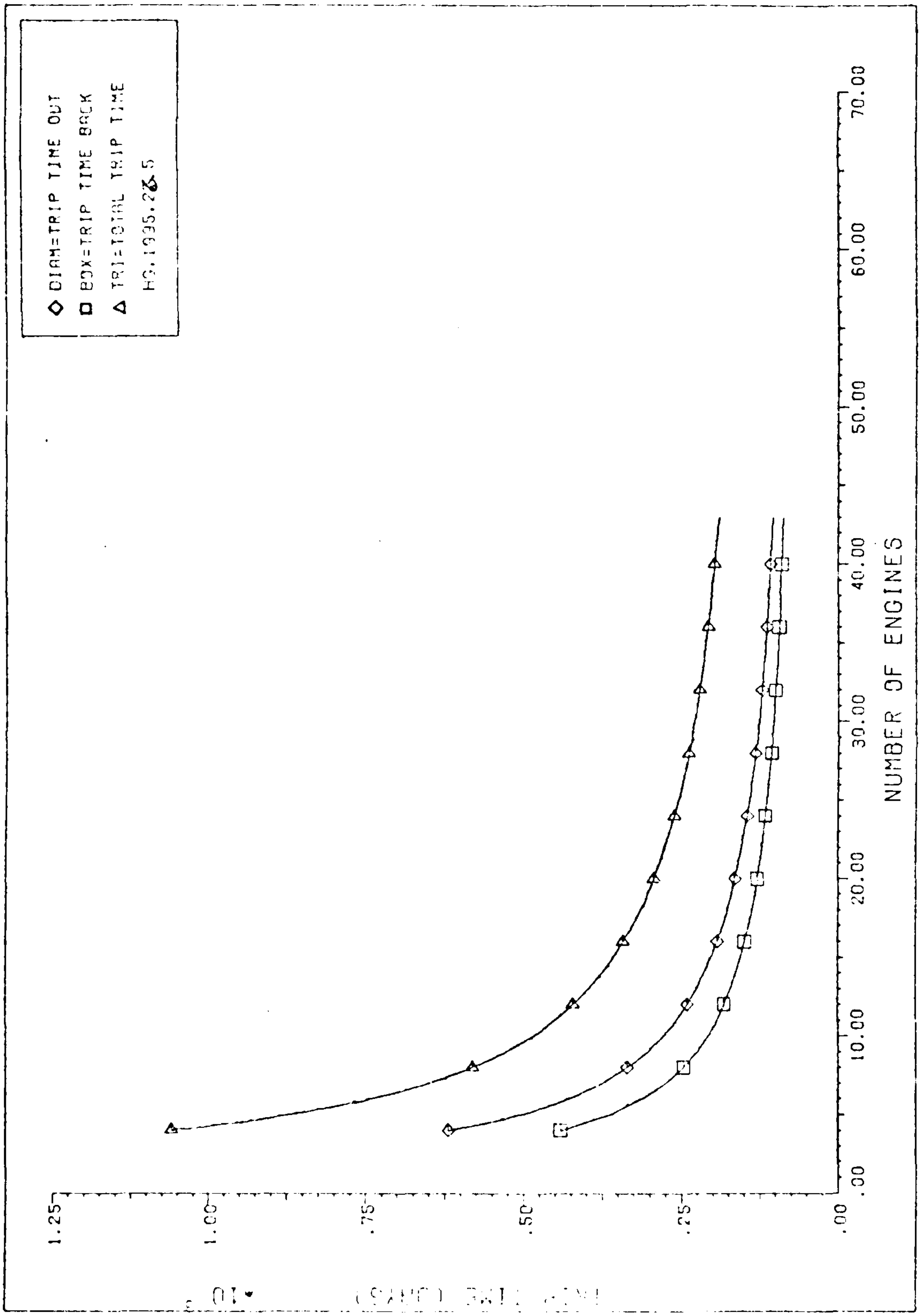
COST VS. NUMBER OF ENGINES HG,AR,XE,1995,26.5

Figure E-4



COSTS VS. NUMBER OF ENGINES HG.1995.26.5

Figure E-5



TRIP TIME VS. NUMBER OF ENGINES HG.1935.26.5

Figure E-6

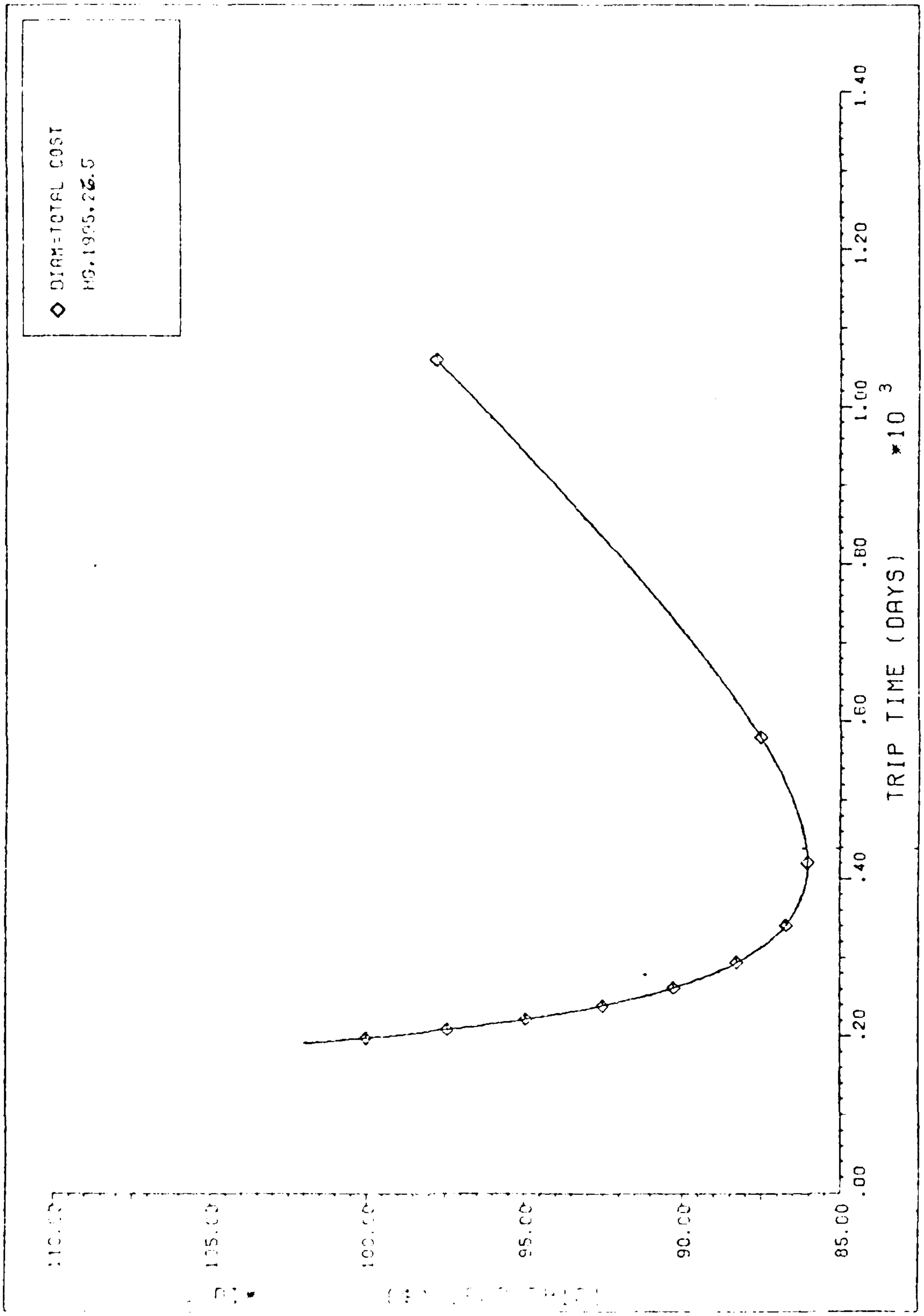

DIAM=TOTAL COST
TRIP TIME (DAYS)
*10 3
.00
.20
.40
.60
.80
1.00
1.20
1.40
85.00
90.00
95.00

Figure E-7

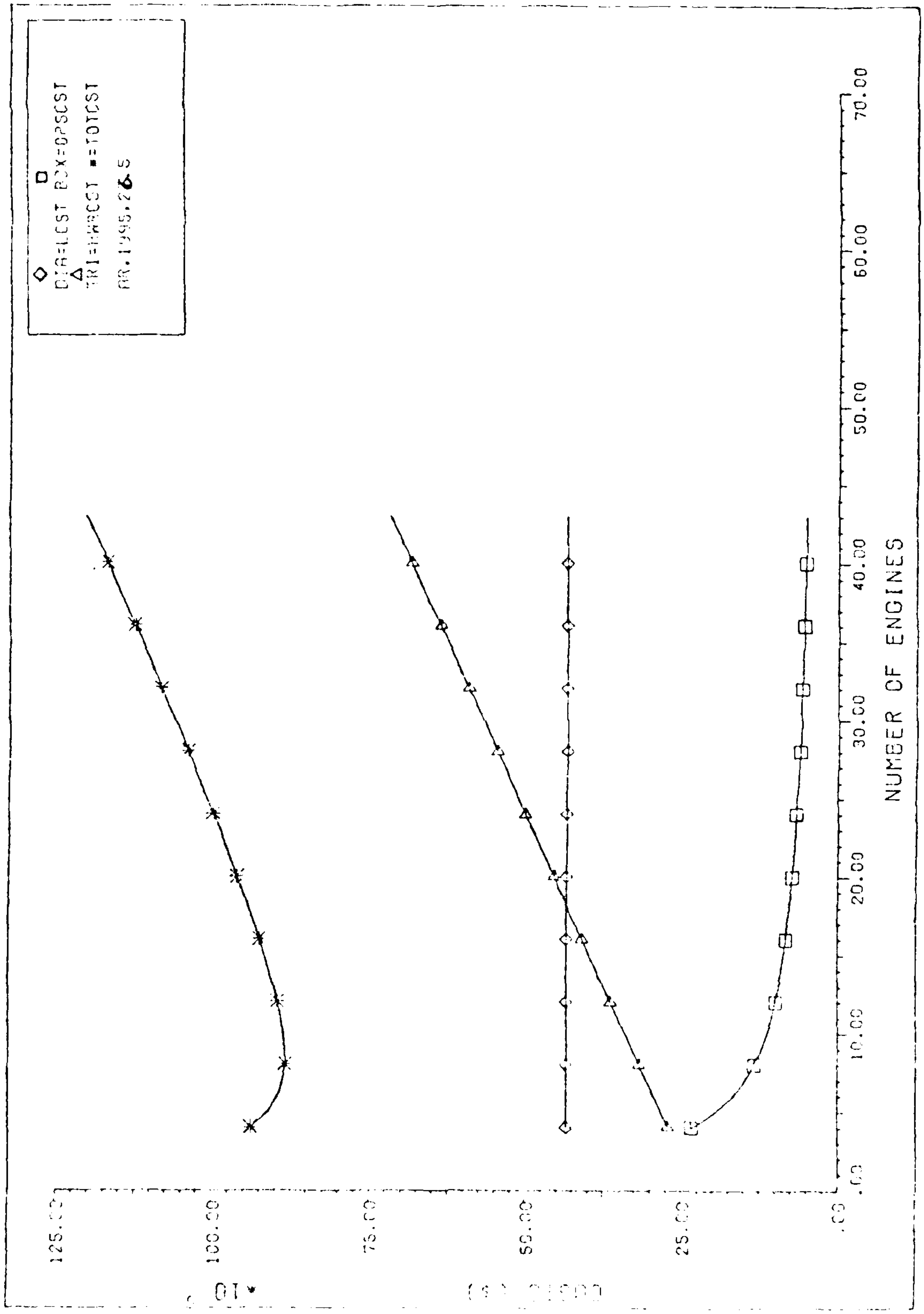
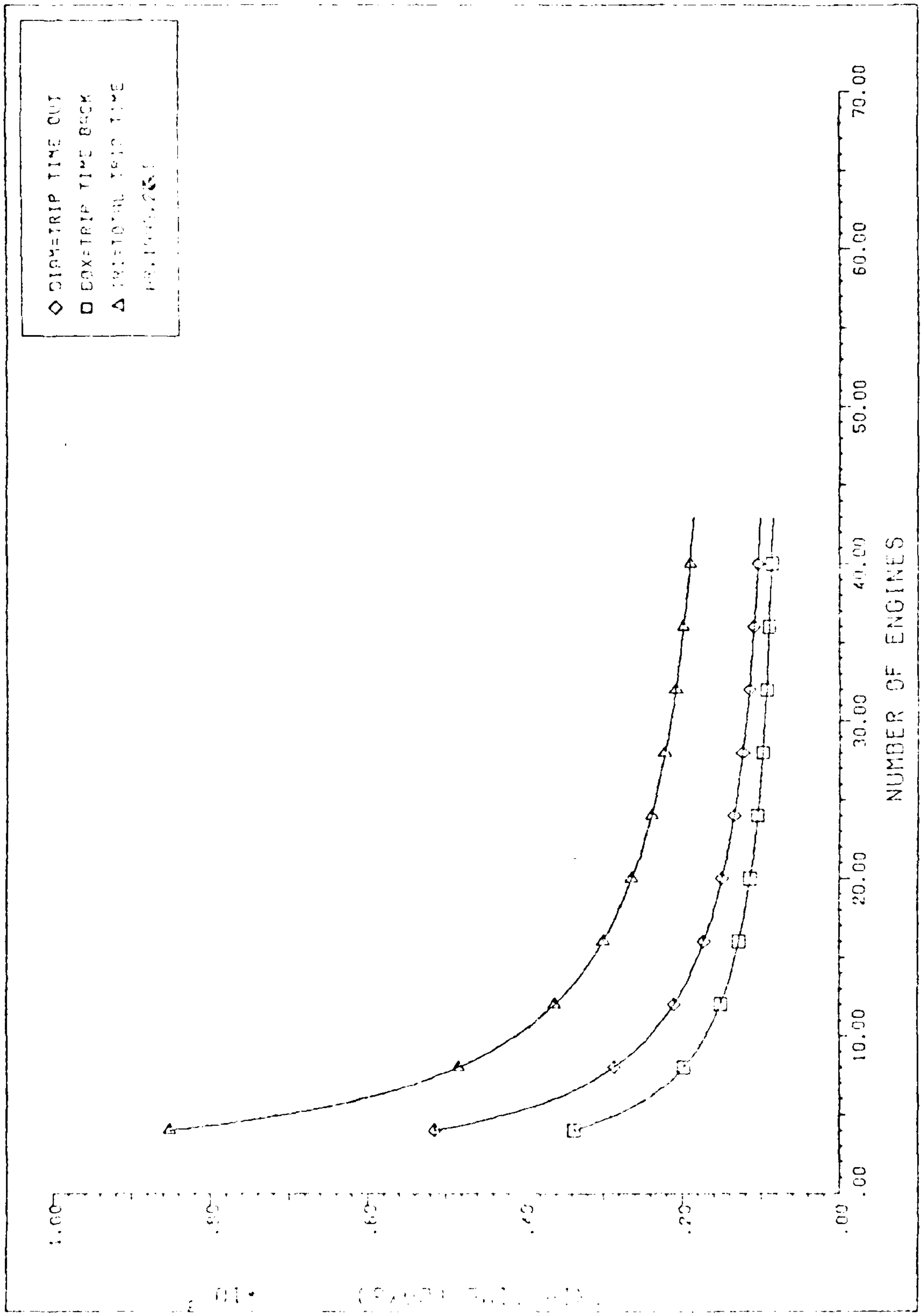

NUMBER OF ENGINES
.00
10.00
20.00
30.00
40.00
50.00
60.00
70.00
.00
25.00
50.00
75.00
100.00
125.00

Figure E-8



TRIP TIME VS. NUMBER OF ENGINES [illegible]

Figure E-[illegible]

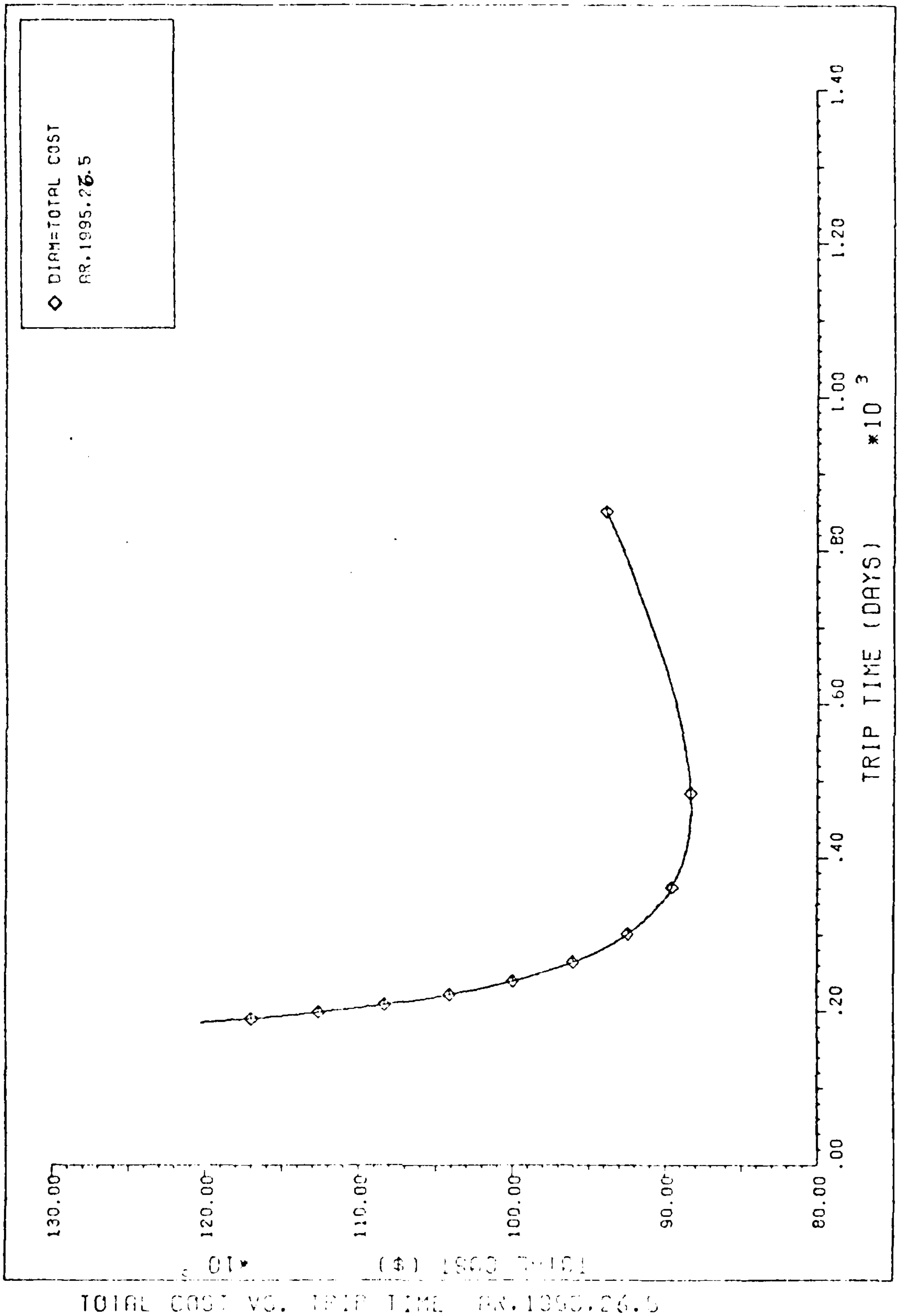
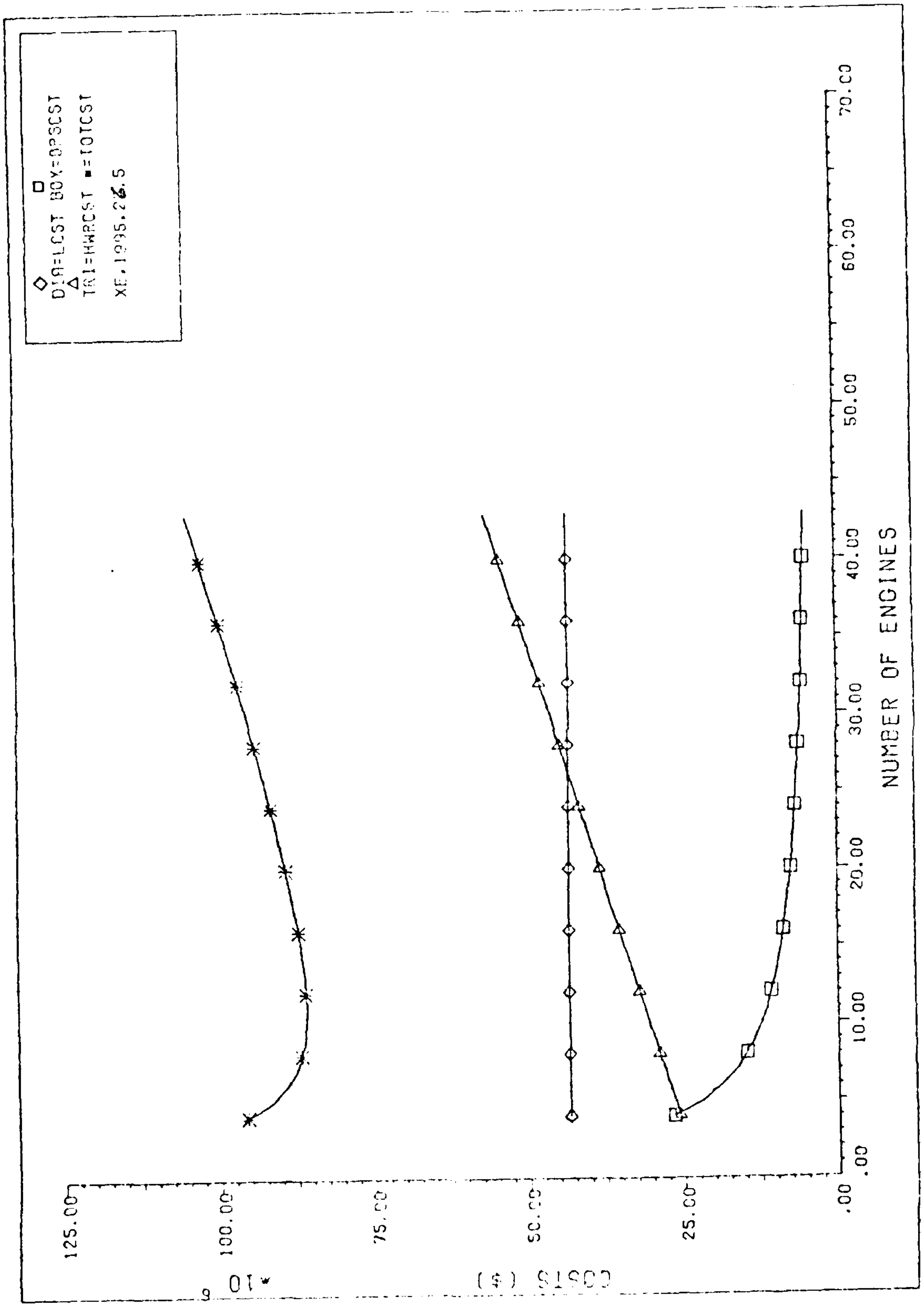
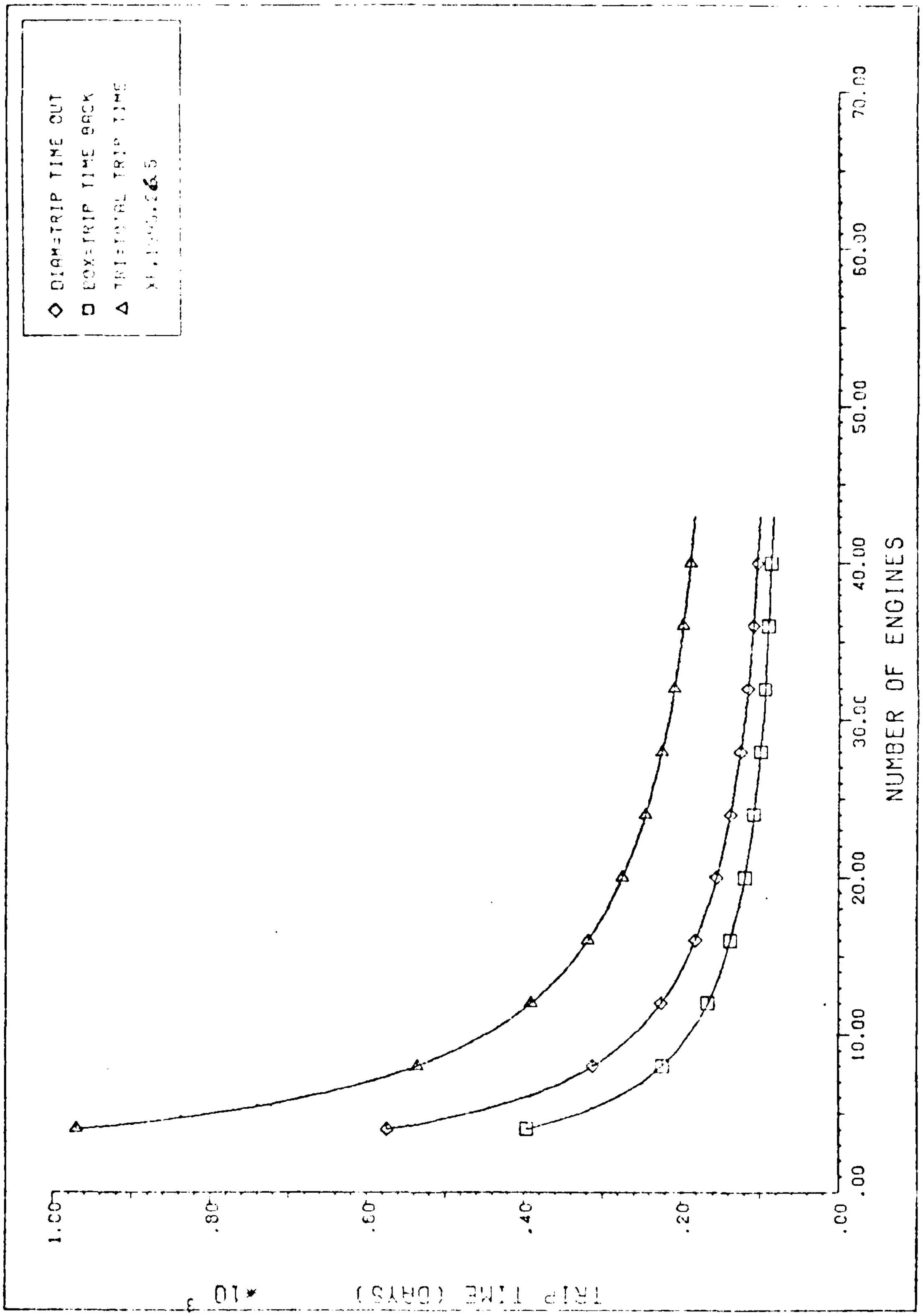

◇ DIAM=TOTAL COST
AR.1995.26.5
TRIP TIME (DAYS)
*10 3
.00
.20
.40
.60
.80
1.00
1.20
1.40
130.00
120.00
110.00
100.00
90.00
80.00

Figure E-10



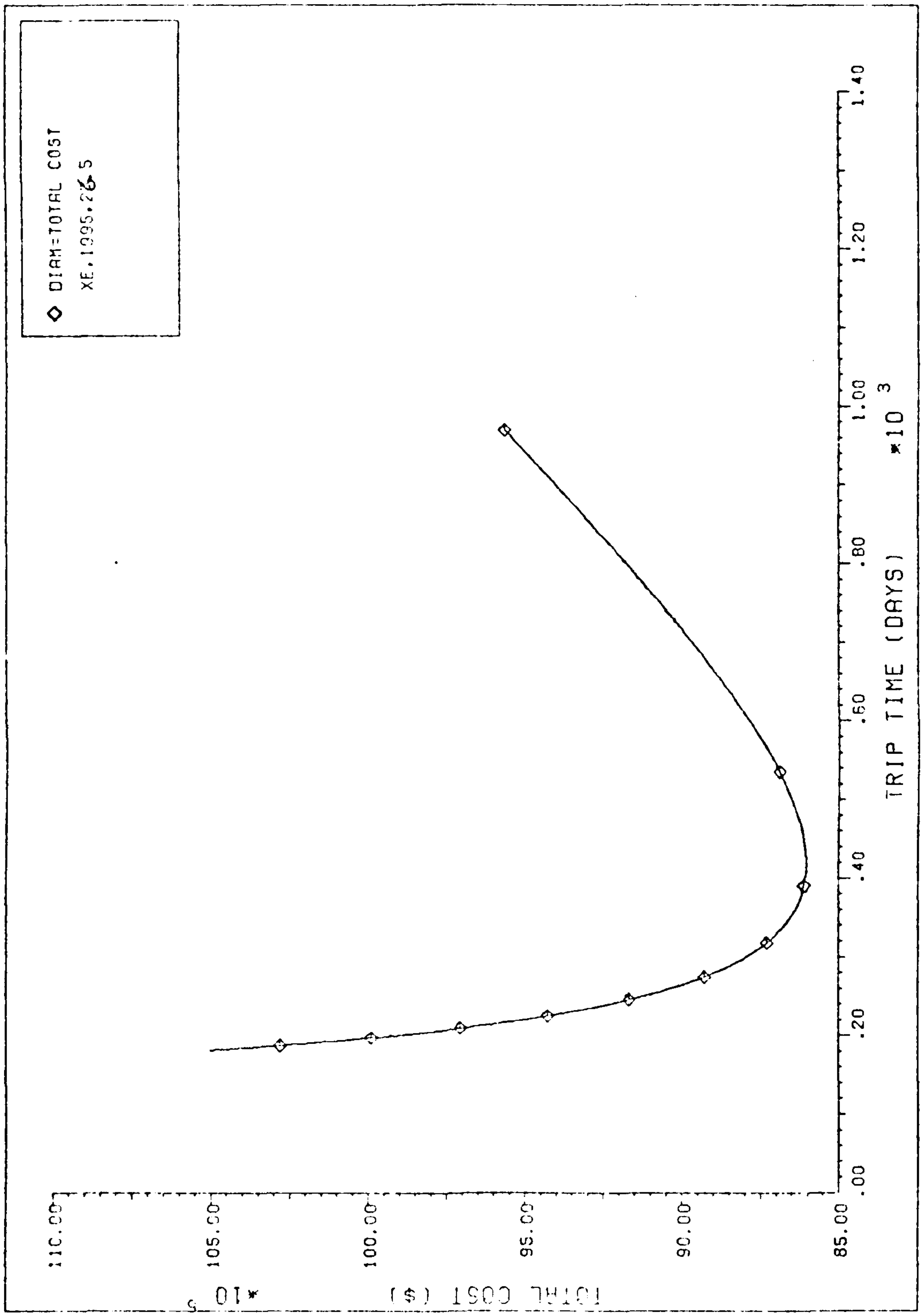
COSTS VS. NUMBER OF ENGINES XE.1995.26.5

Figure E-11



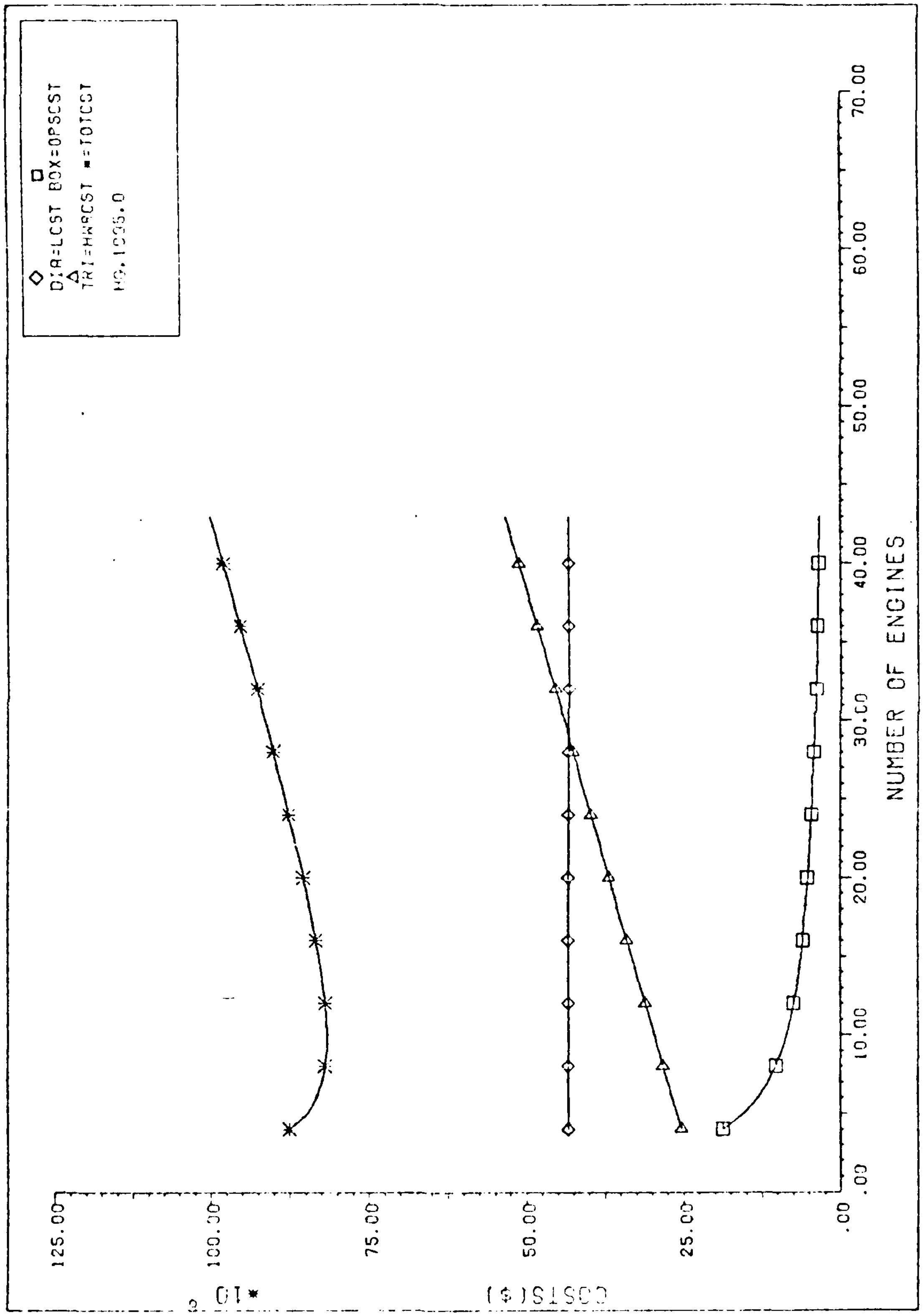
TRIP TIME VS. NUMBER OF ENGINES XE.1975.26.5

Figure E-12



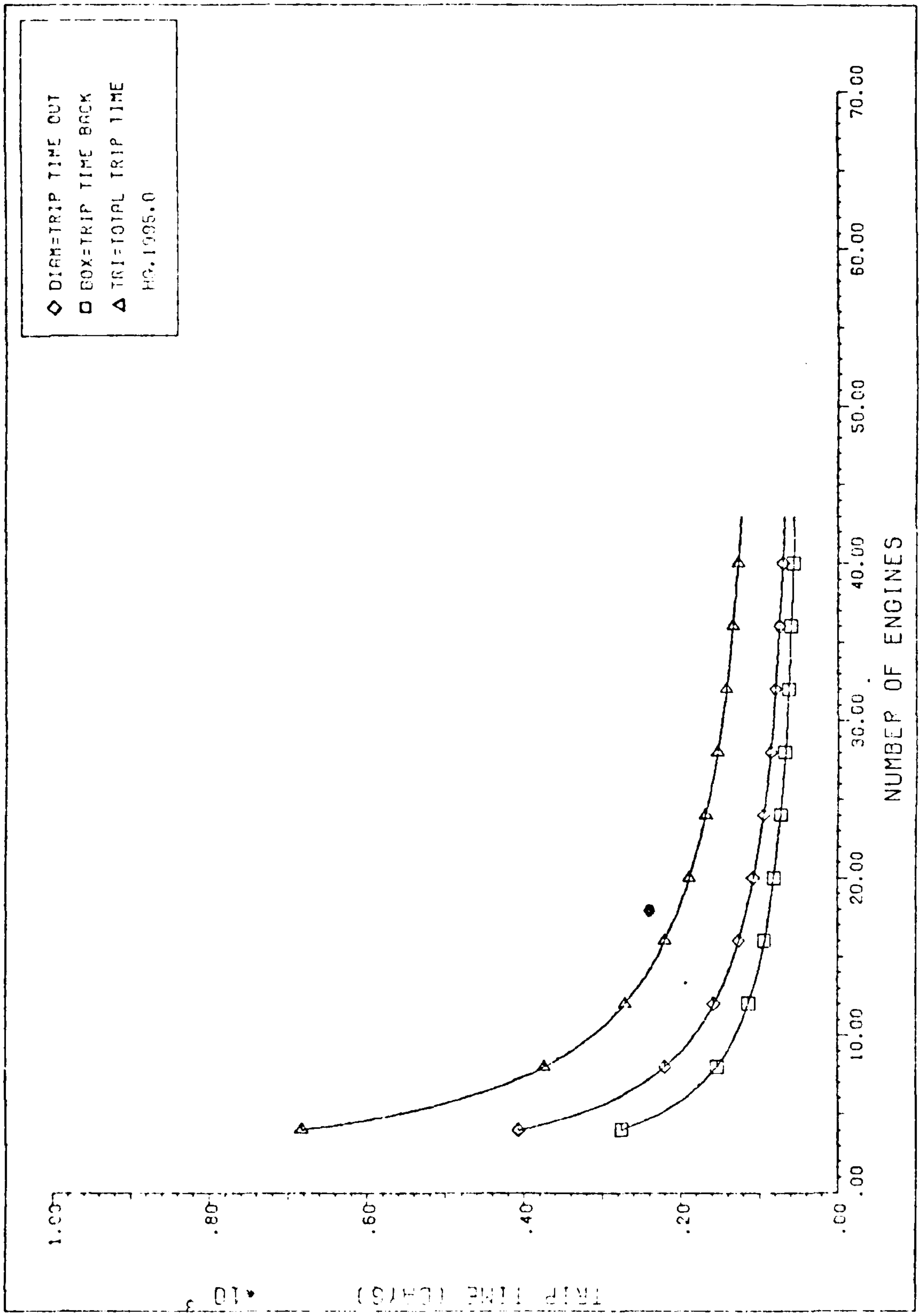
TOTAL COST VS. TRIP TIME XE.1995.26.5

Figure E-13



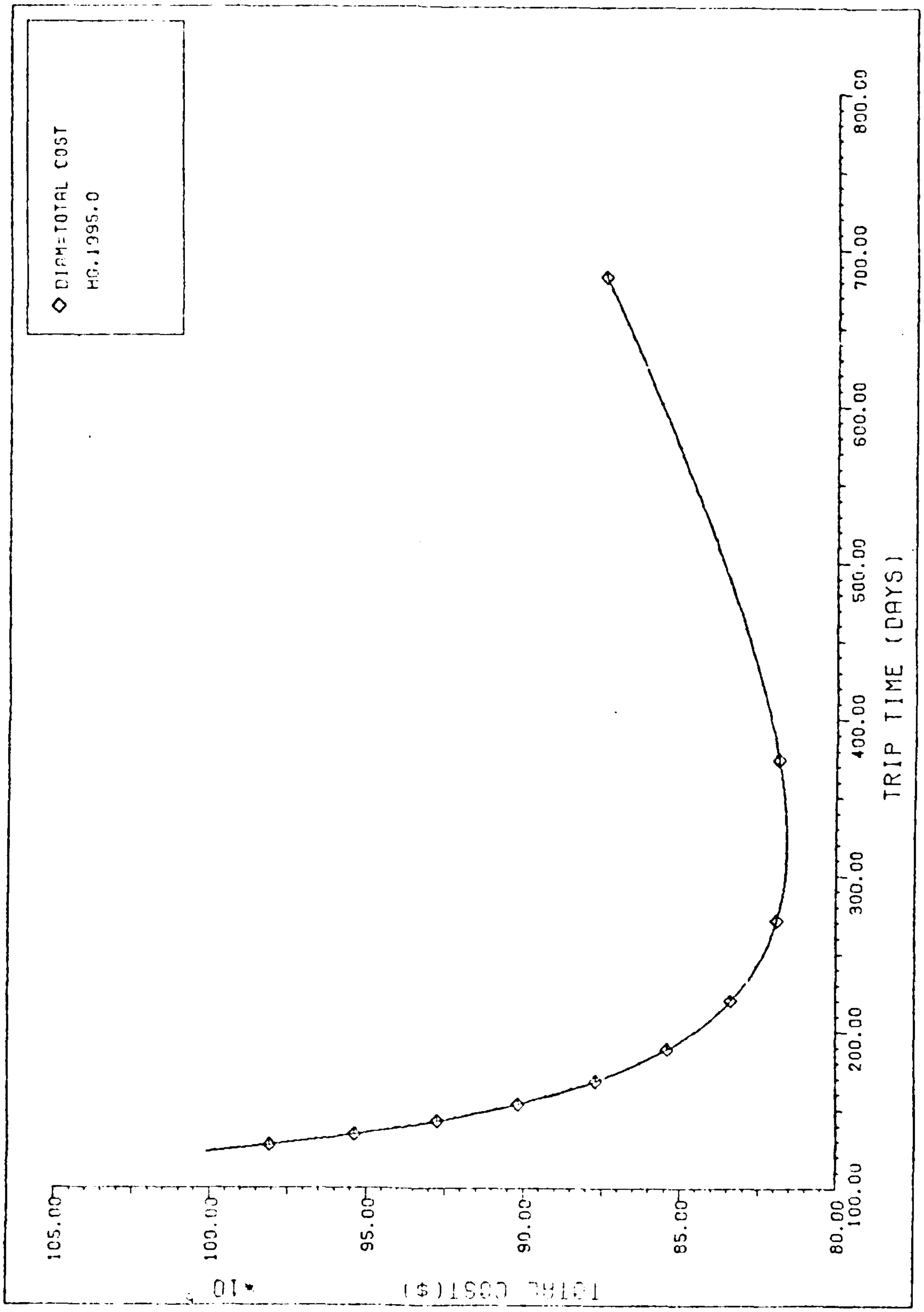
COSTS VS. NUMBER OF ENGINES HG.1995.0

Figure E-14



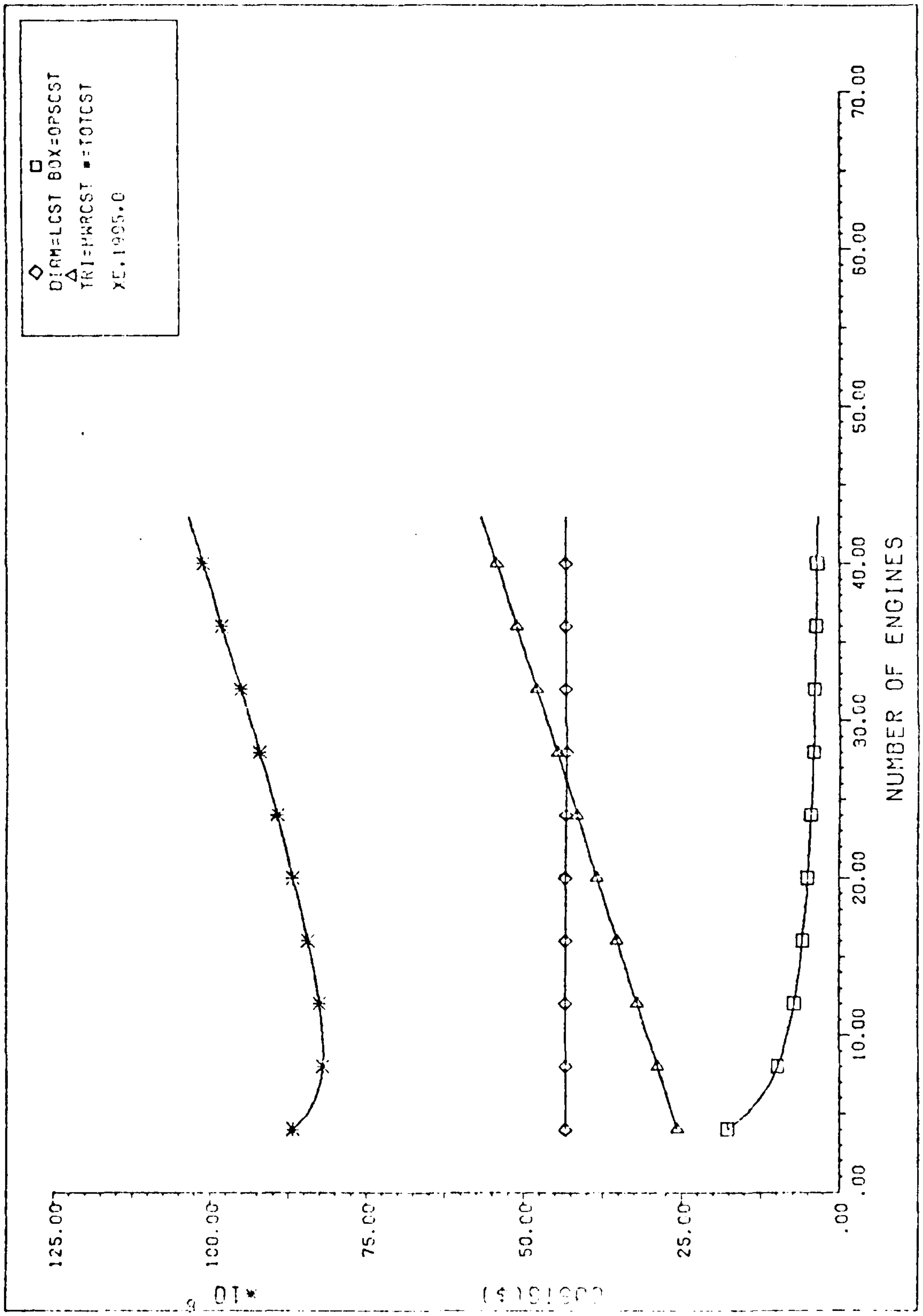
TRIP TIME VS. NUMBER OF ENGINES HG.1995.0

Figure E-15



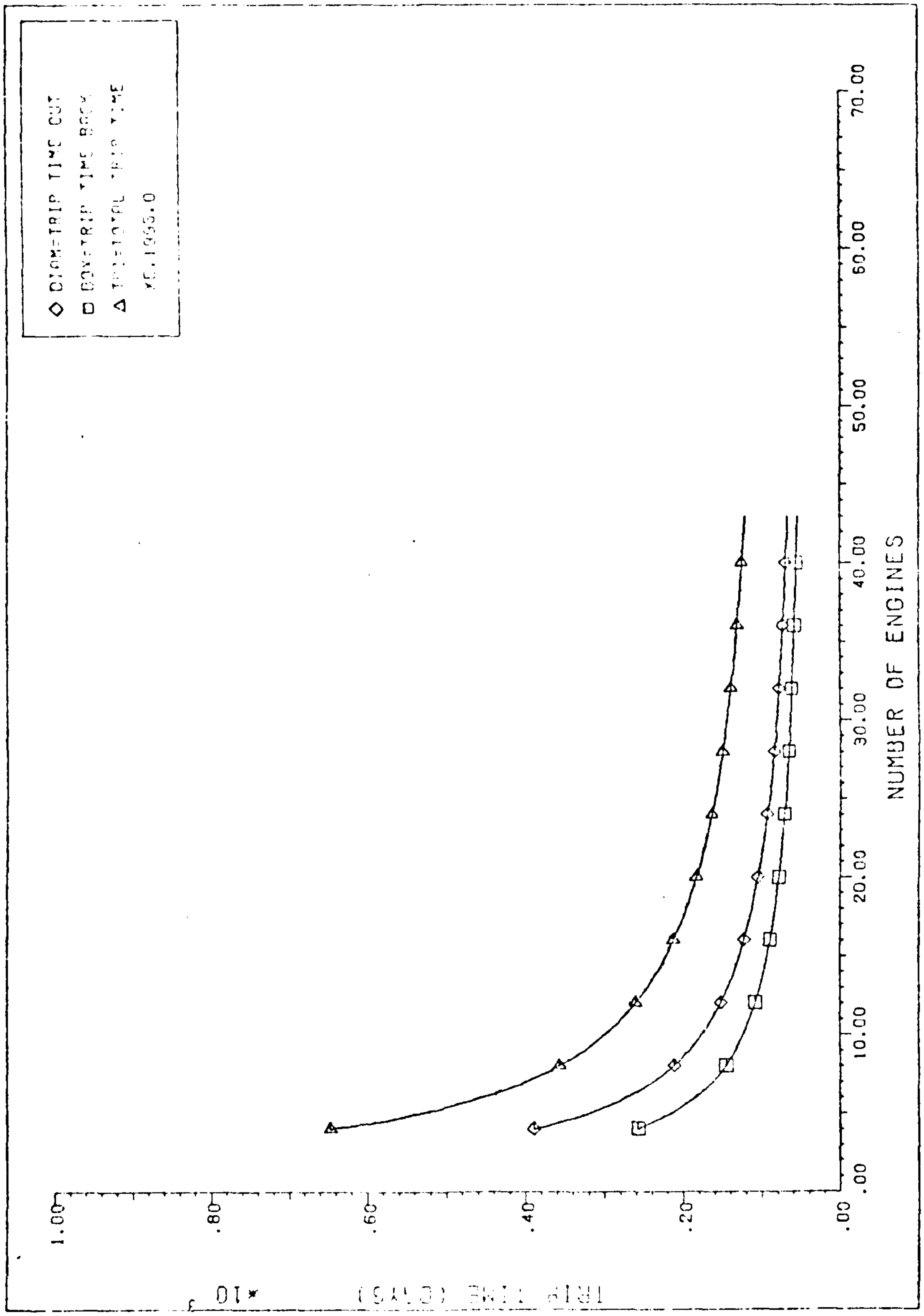
TOTAL COST VS. TRIP TIME HG.1995.0

Figure E-16



COSTS VS. NUMBER OF ENGINES XE.1995.0

Figure E-17



TRIP TIME VS. NUMBER OF ENGINES XE.1985.0

Figure E-18

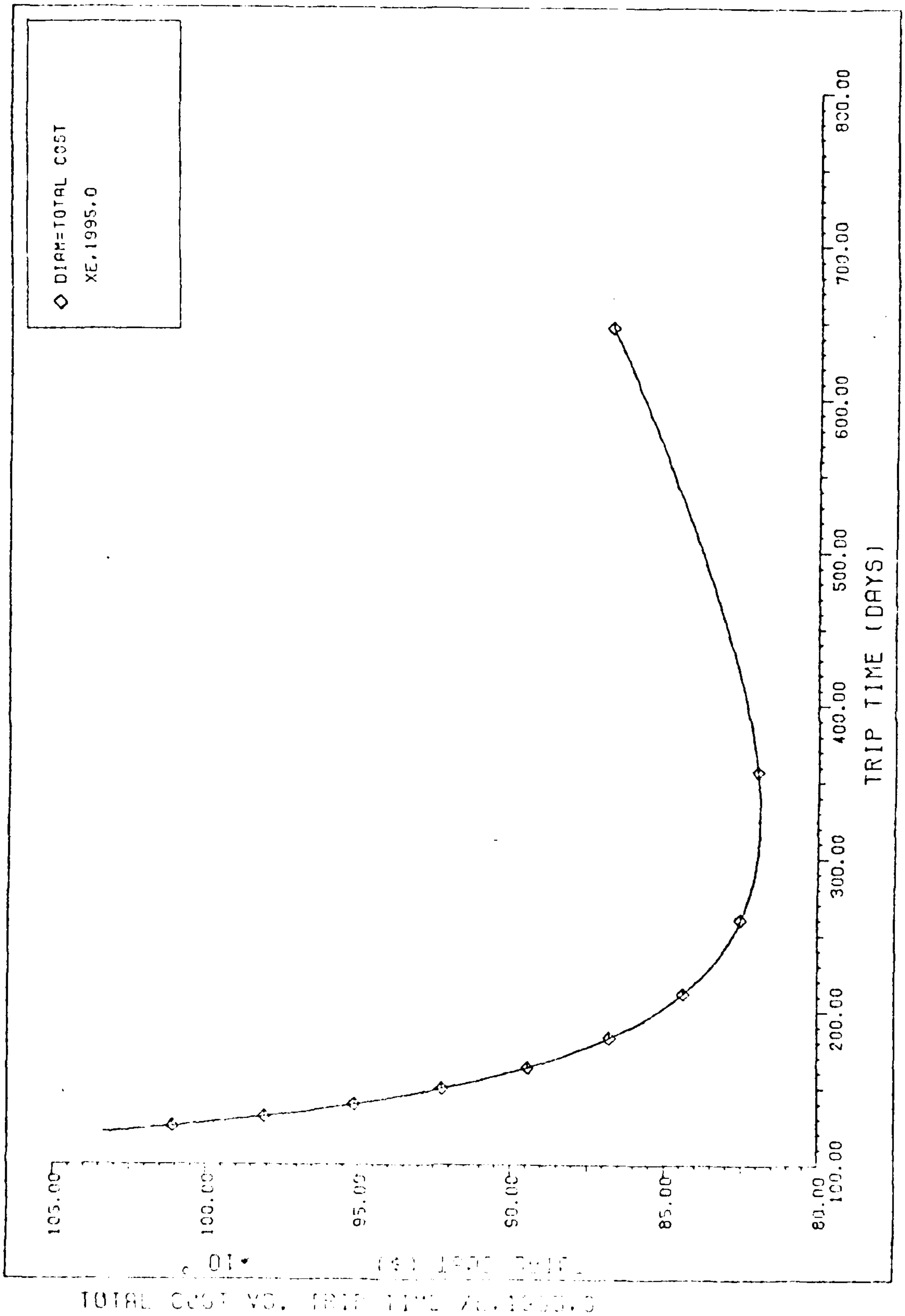
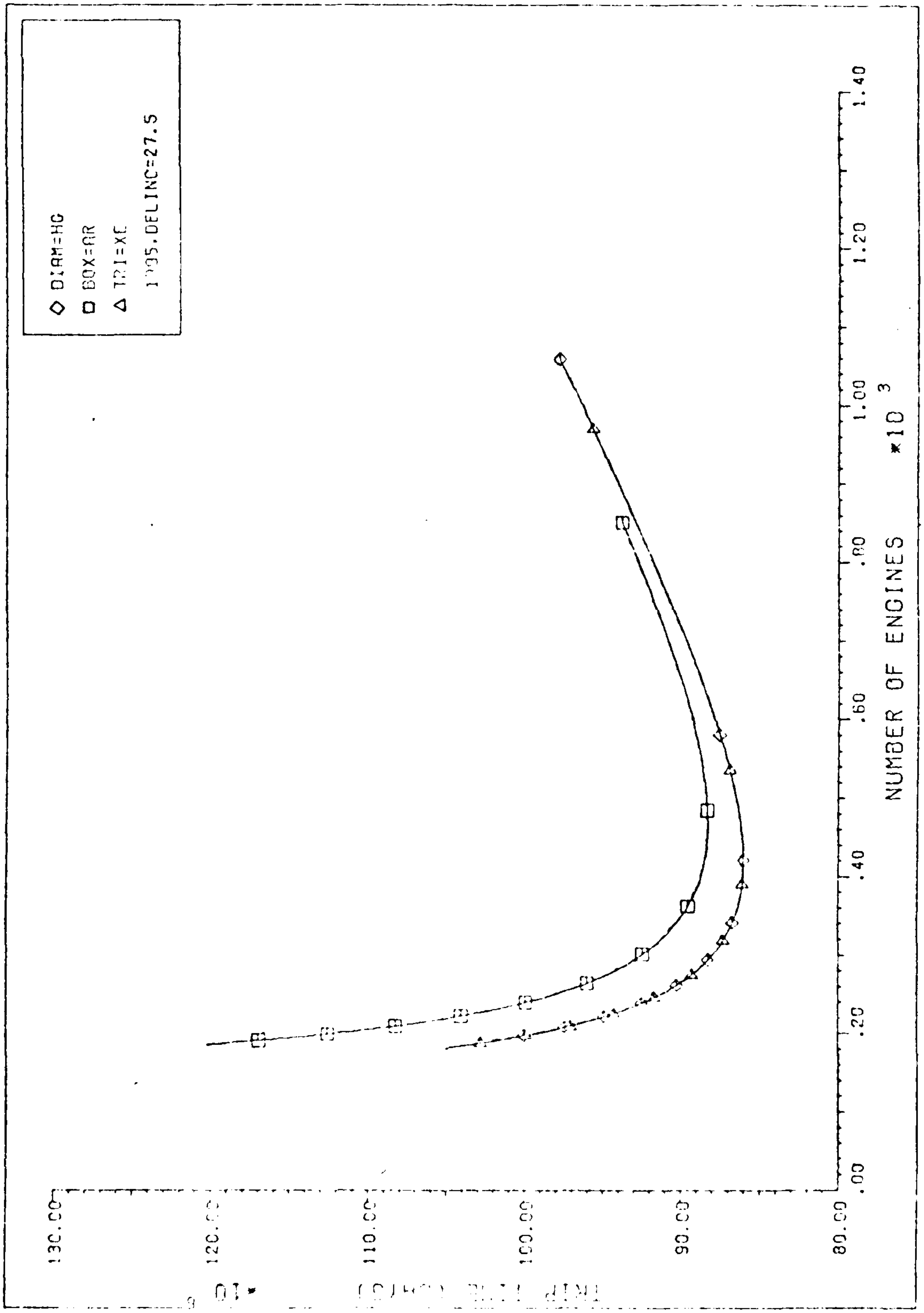
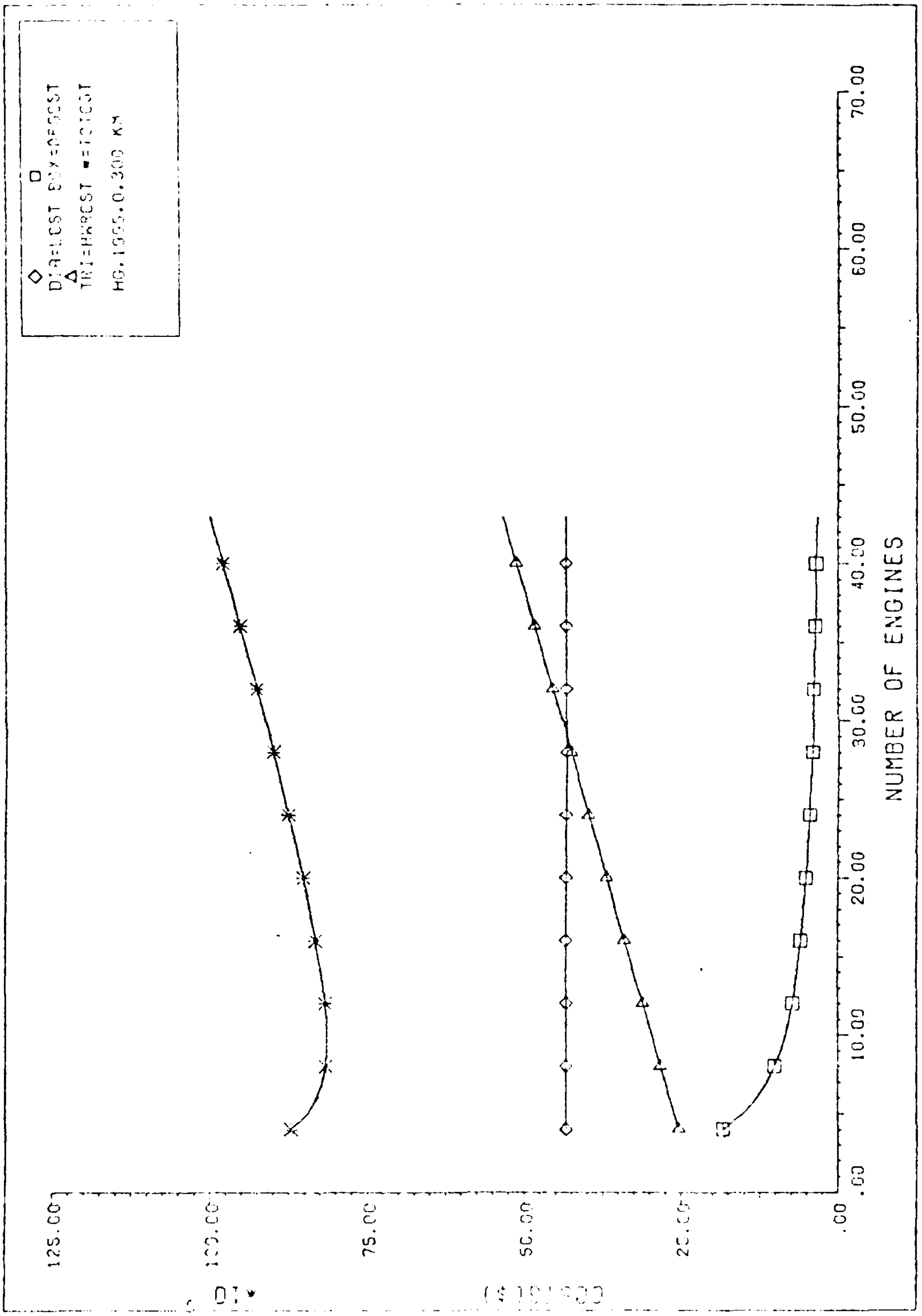

DIAM=TOTAL COST
XE,1995.0
TRIP TIME (DAYS)
100.00
200.00
300.00
400.00
500.00
600.00
700.00
800.00
105.00
100.00
95.00
90.00
85.00
80.00
TOTAL COST VS. TRIP TIME XE,1995.0

Figure E-19



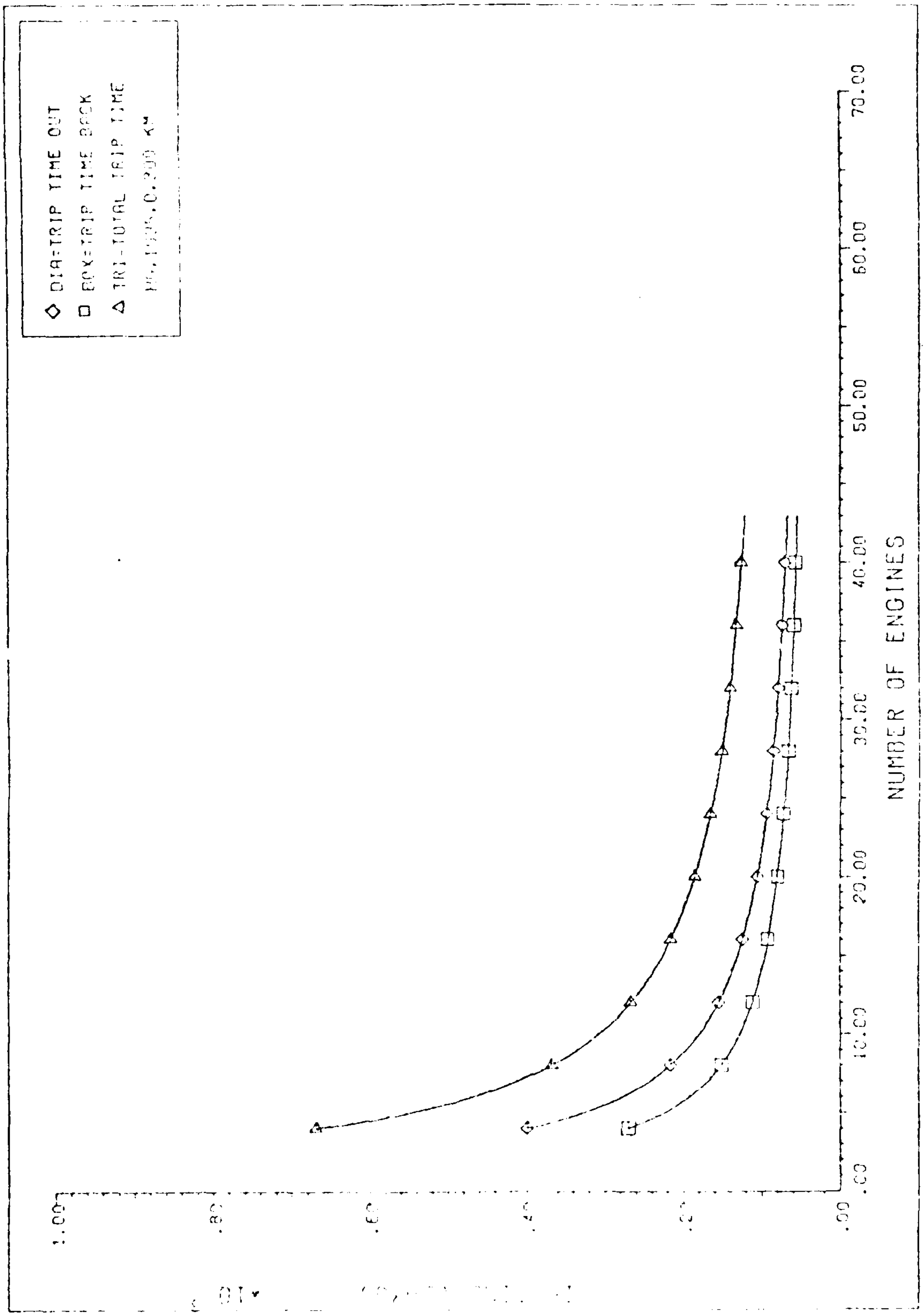
TRIP TIME VS. NUMBER OF ENGINES HG,AR,XE,1995.0

Figure E-20



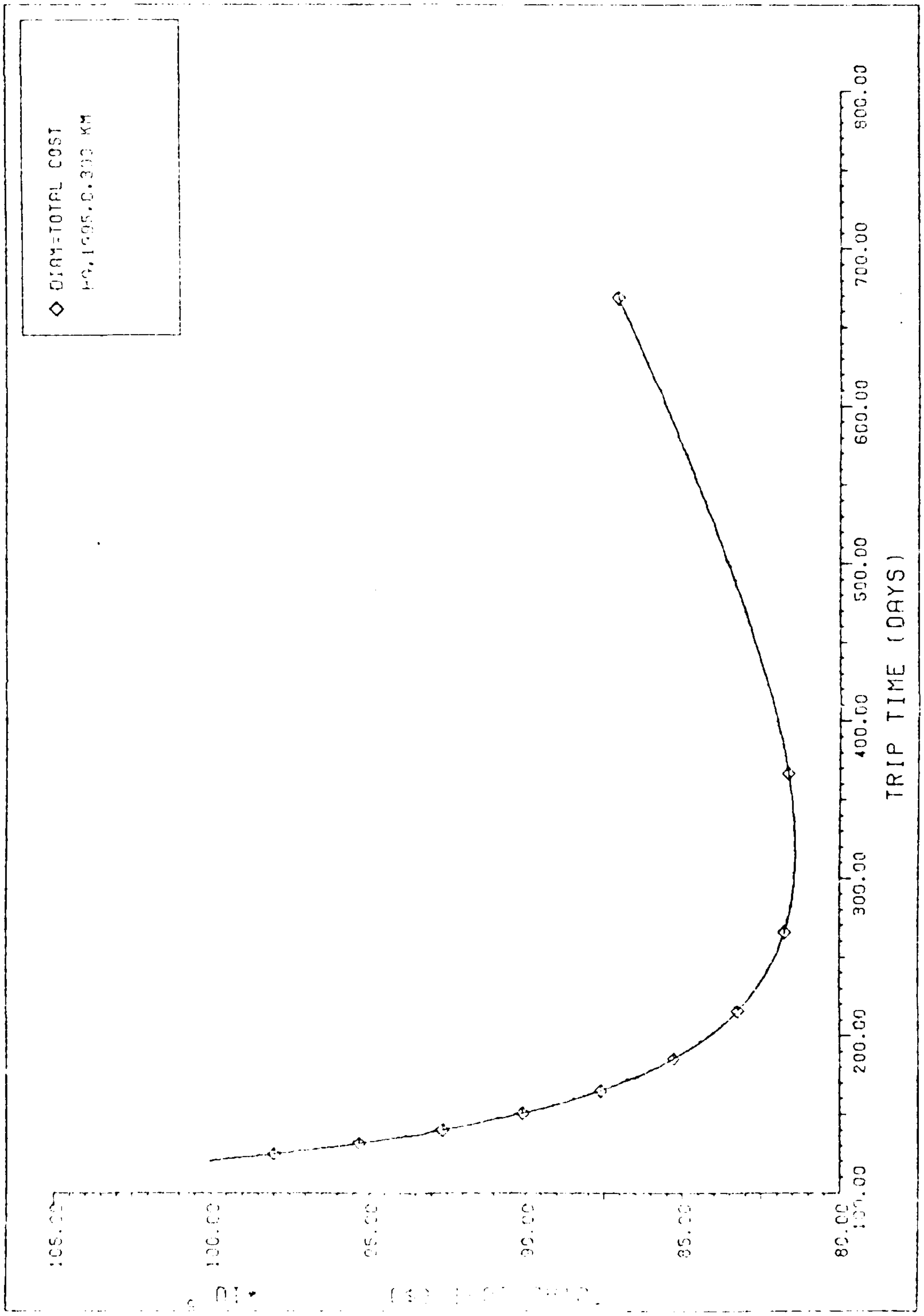
COSTS VS. NUMBER OF ENGINES HG.1995.0.300 KM

Figure E-21



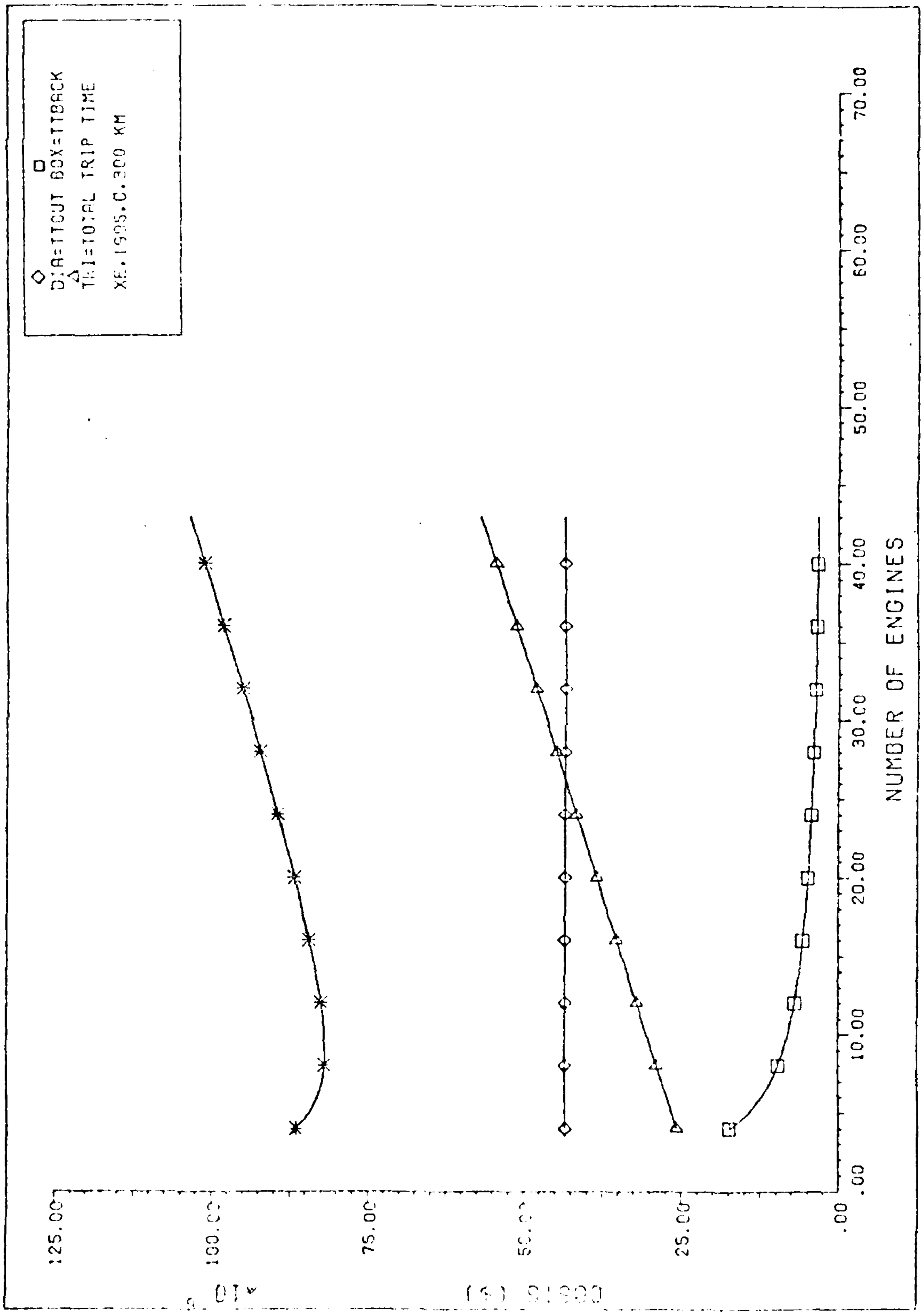
TRIP TIME VS. [illegible]

Figure E-2[illegible]



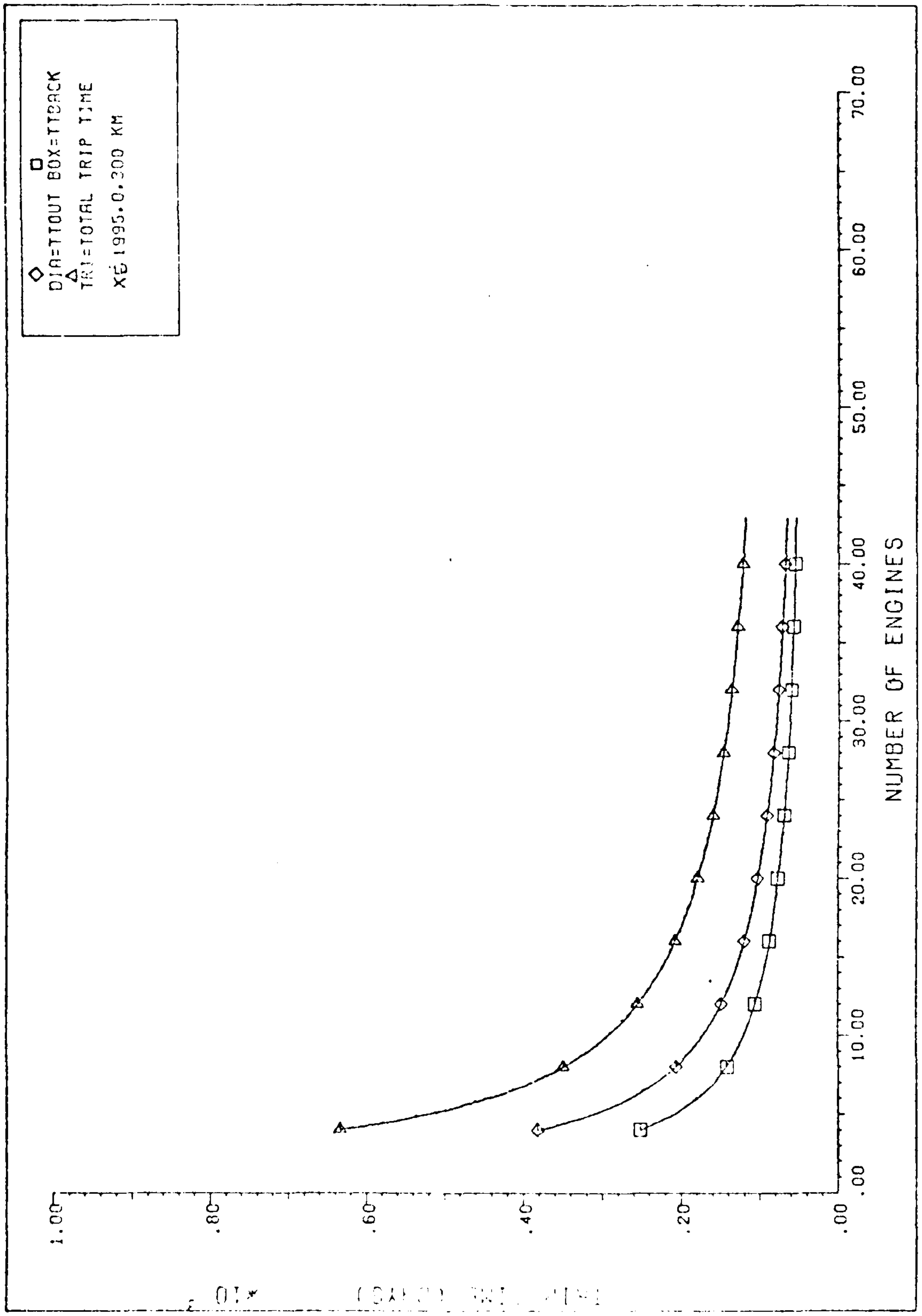
TOTAL [illegible]

Figure E-13



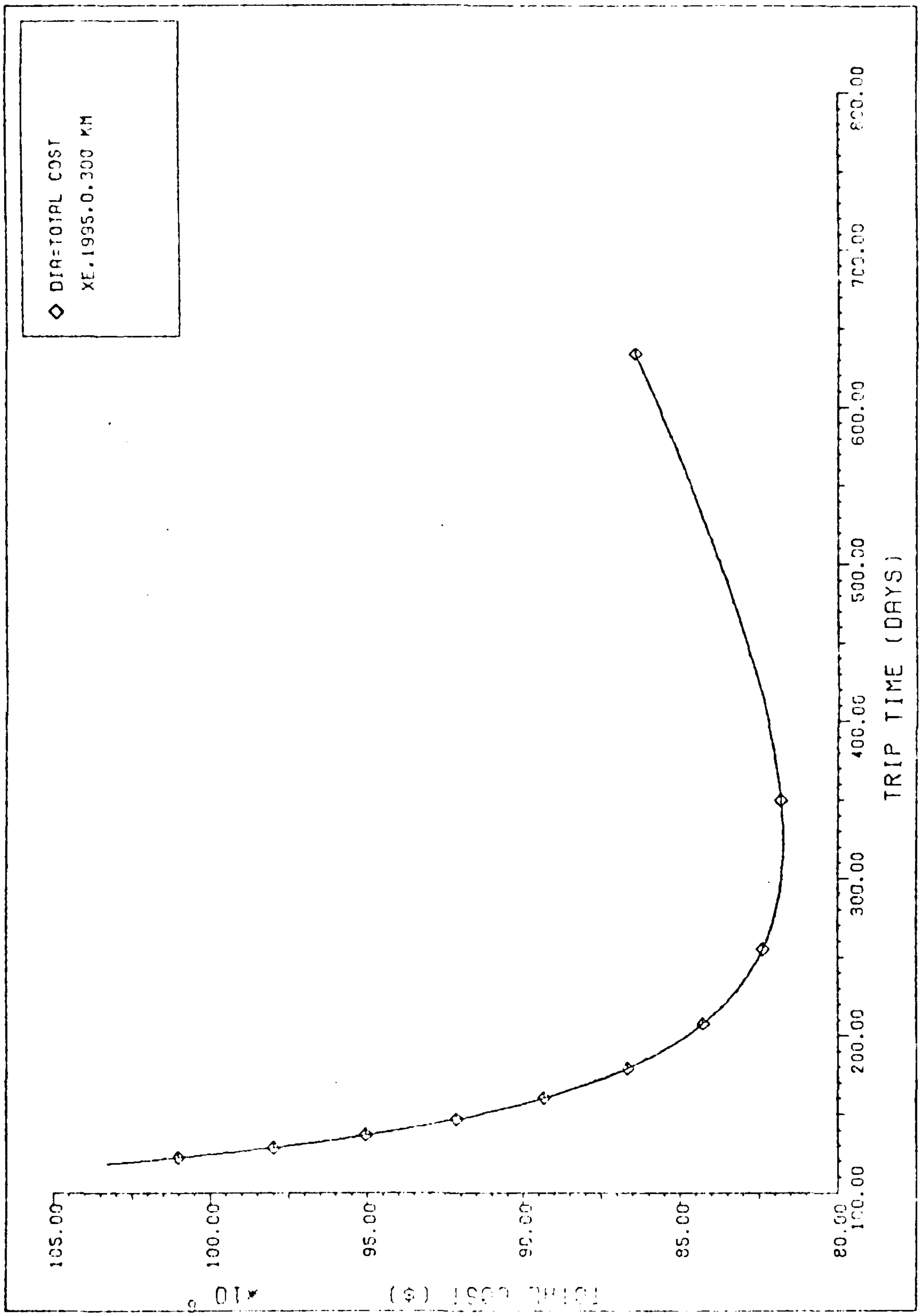
COSTS VS. NUMBER OF ENGINES XE.1995.C.300 KM

Figure E-24



TRIP TIME VS. NUMBER OF ENGINES [illegible] 300 KM

Figure E-2[illegible]



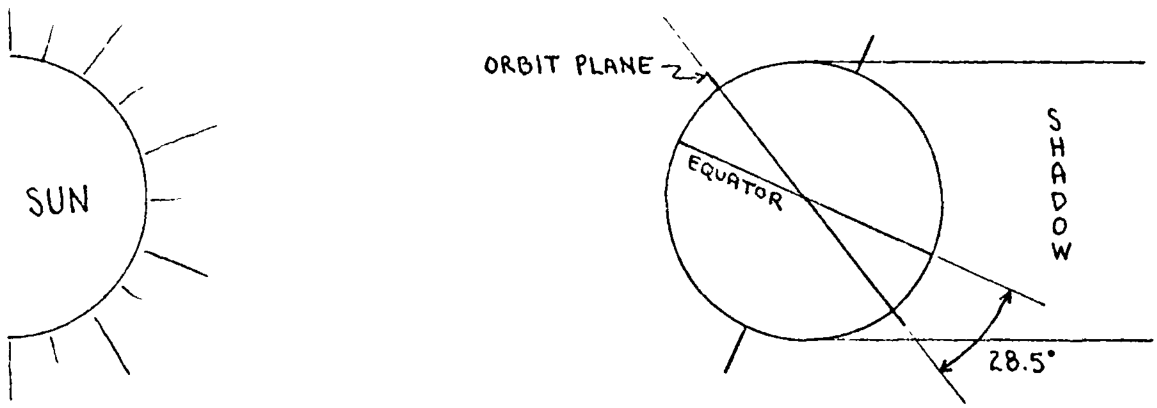
TOTAL COST VS. TRIP TIME XE.1995.0.300 KM

Figure E-26

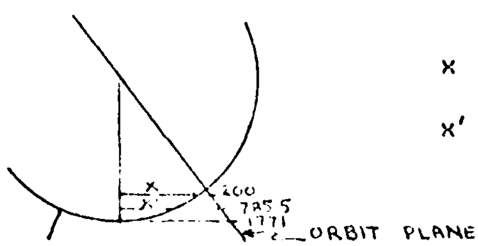
APPENDIX F

DERIVATION OF VALUES FOR PHI AND TD

The calculations shown here are strictly for obtaining first order estimates of the penalty factor to account for the time the EOTV is in the earth's shadow (PHI) and of the penalty factor for engine restart (TD). The orientation of the initial orbit is dependent on the time of launch of the Shuttle. It is possible to select the launch time to produce an orbit that is oriented to give the least amount of time in the earth's shadow. This is illustrated below :



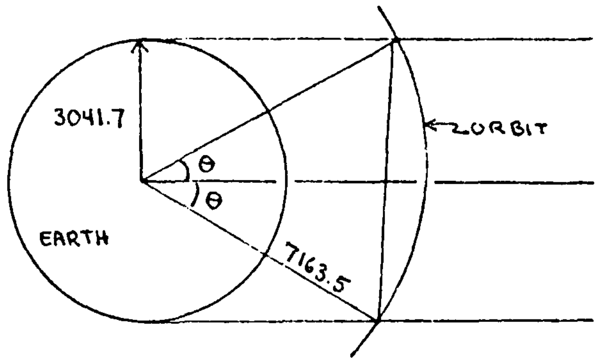
Using simple geometry it was possible to determine that the orbit altitude would have to be increased to 1771 km in order to escape the earth's shadow completely. As the orbit altitude is increased, the time in the shadow decreases because the radius of the earth producing the shadow decreases as illustrated below.



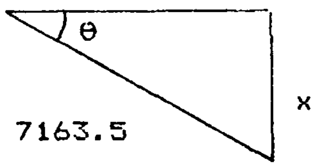
$x = 3765$ km

$x' = 3042$ km

For an altitude of 785.5 km - midway between 200 km and 1771 km - the radius of the earth that casts a shadow across the orbit plane is 3041.7 km. At this altitude, the orbit period is 100.6 minutes and the portion of the orbit in the shadow is 14.03 minutes. This is derived below :



(Top View)
(not drawn to scale)



x = 3041.7

$$\theta = \sin^{-1}\left(\frac{3041.7}{7163.5}\right) = 25.1$$

$$2\theta = 50.2$$

$$\frac{2\theta}{360} = .139$$

$$.139 \times 100.6 = 14.03$$

It will take the EOTV approximately 20 days to increase its orbit altitude to 1771 km (based on a 110 day total transfer time). In this time, approximately 286 orbits will be made and a total of 66.9 hours (2.79 days) will be spent in the shadow. This time in the earth's shadow is .025 of

the total trip time and this value was used for PHI in the model.

The calculation of the penalty for engine restart (TD) is much simpler. According to Mr. Rawlin of the NASA Lewis Research Center, it takes 18 minutes to get a cold engine to full power. The engines will be off only when in the earth's shadow and useful thrust will be lost only when the EOTV emerges from the shadow. Using the same orbit period of 100.6 minutes as above, this 18 minute delay corresponds to .18 of the orbit. This is the value used for TD in the model.

APPENDIX G

DERIVATION OF SOLAR ARRAY SPECIFIC MASS AND SPECIFIC POWER

The values used for the solar array specific mass (kg/KW) and the specific power (W/sq ft) were derived from data on the Solar Electric Propulsion System (SEPS) Arrays contained in Reference 57. From this data it was possible to derive an equation that gives the weight per square foot of an array ($\bar{W}$) depending on the type of solar cell and thickness of cover used. This equation is shown below :

$$\bar{W} = .16862 + .0051915\, k_p \,(\rho_{cell}\, t_{cell} + \rho_{cover}\, t_{cover})$$

where

$\bar{W}$ is in lbs/sq ft

.16862 represents the structure weight

.0051915 is a units conversion factor

k_p (packing factor) = .93

ρ_{cell} (density of the solar cell in gm/sq cm) = 2.33 for Si

t_{cell} (cell thickness in mils)

ρ_{cover} (density of the shielding in gm/sq cm) = 2.51 for microsheet

t_{cover} (shielding thickness in mils)

The specific power ($\bar{P}$) in W/sq ft was determined using :

$$\bar{P} = k_p\, k_t\, k_r\, k_w\, \eta_o\, H_o$$

where

k_p (packing factor) = .93

k_t (thermal loss or gain factor) = 1.13 for Si

k_r (radiation loss factor) = 1 for BOL

k_w (wiring loss factor) = .97

η_o (cell efficiency) = .135 for Si

H_o (solar constant) = 125.6978 W/sq ft

The value for specific power is constant for a given array type and is independent of shield thickness. For an array of four mil silicon cells, $\bar{P}$ is equal to 17.298 W/sq ft.

The values for the solar array specific mass (ASA) were obtained by simply dividing $\bar{P}$ by $\bar{W}$ (with appropriate unit conversions). The values obtained using 4 mil silicon arrays are :

Shield thickness (mils)	$\bar{W}$ (kg/sq ft)	ASA (kg/KW)
3	.0968954	6.873
6	.1298767	7.508
12	.1628578	9.415
20	.2068329	11.957

APPENDIX H

DEPLOYMENT COSTS FOR THE CHEMICAL OTVS

The calculation of deployment costs for the chemical OTVs is relatively simple. Aside from the purchase price for the upper stage itself, the only other cost included is the launch cost to get the OTV and payload into LEO. The guidance and control costs during the orbit transfer are omitted because the orbit transfer takes only six hours.

In calculating the total payload mass (MT) for the shuttle, a factor of .125 is included to account for shuttle adaptive hardware. The shuttle launch costs (CETO) are computed using the equations listed in Chapter 5. In all cases, the cost using the weight factor was greater than the cost using the lenght factor. Only the wieght factor cost is presented below.

The information below includes :

- OTV weight

- OTV maximum payload to GEO. The delta V required for GEO is only slightly higher than the delta V required for the GPS orbit; therefore, the payload capability to the GPS orbit would be slightly higher.

- MT (total shuttle payload weight)

- CETO (launch cost to LEO)

- Purchase price for the OTV

- Total deployment cost for one satellite

PAM D-II

Weight :	5566 kg
Max. payload :	1842 kg
MT :	7949 kg
CETO :	$ 23.37 Million
OTV cost :	$ 6 - 10 Million
Total cost :	$ 29.37 - 33.37 Million

CENTAUR-G

Weight :	16329 kg
Max. payload :	4808 kg
MT :	20058 kg
CETO :	$ 58.96 Million
OTV cost :	$ 30 Million
Total cost :	$ 88.96 Million

IUS

Weight :	14656 kg
Max. payload :	2722 kg
MT :	18176 kg
CETO :	$ 53.43 Million
OTV cost :	$ 84 Million
Total cost :	$ 137.4 Million

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VITA

Captain David P. Boyarski was born on 5 March 1956 in Cebu City, Philippines. He graduated from high school in Cebu City in 1972. He came to the United States in June 1973 and entered the United States Air Force Academy from which he received the degree of Bachelor of Science in Engineering Sciences in June 1977. Upon graduation, he entered pilot training at Vance AFB, Oklahoma. He completed pilot training and received his wings in August 1978. He then served as a C-141 pilot, instructor, and flight examiner in the 15th Military Airlift Squadron, Norton AFB, California until entering the School of Engineering, Air Force Institute of Technology in June 1983.

Permanent address: 2925 Downhill Drive
Colorado Springs, CO
80918

VITA

Captain Stephen P. Mahoney was born on 17 March 1954 in Washington, D. C.. He graduated from high school in Springfield, Virginia, in 1972 and attended the United States Air Force Academy from which he recieved the degree of Bachelor of Science in Computers in June 1976. Upon graduation, he was assigned to Columbus AFB, Mississippi, for pilot training, where he recieved his wings in November 1977. He then served as a C-130 pilot in the 32nd and 50th Tactical Airlift Squadrons, Little Rock AFB, Arkansas, until entering the School of Engineering, Air Force Institute of Technology, in June 1983.

Permanent address: 7909 Ellet Road
Springfield, Virginia 22151

Fwd
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